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– Exchanging of scientific and technical information;

– Continuously stimulating advances in the aerospace sciences relevant to strengthening the common defence posture;

– Improving the co-operation among member nations in aerospace research and development;

– Providing scientific and technical advice and assistance to the North Atlantic Military Committee in the field of aerospace research and development;

– Rendering scientific and technical assistance, as requested, to other NATO bodies and to member nations in connection with research and development problems in the aerospace field;

– Providing assistance to member nations for the purpose of increasing their scientific and technical potential;

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PREFACE

The AGARD multi-panel Symposium on Fighter Aircraft Design was held in response to a request from the NATO Military Committee to assess the state of technology as related to future fighter aircraft design. The meeting was initiated by the Flight Mechanics Panel with other panels organising sessions concerned with their particular areas of expertise. The Symposium took place at the Scuola di Guerra Aerea, Florence, Italy on October 3–6, 1977.

The Symposium was organised into nine sessions as follows:–

1. Fighter Requirements for the 80's and Beyond and their Impact for Technology Development.
2. Current Experience.
3. System Design Approach to Fulfil the Requirements.
4. Aerodynamics and Aircraft Configurations.
5. Propulsion.
6. Structures.
7. Avionics/Guidance.
8. Human Factors.
9. Round Table Discussion.

The primary conclusions from the meeting are that technology is available, in all areas of fighter design, to meet the military requirements for the 80's. However, the cost of using the most advanced technology to meet every conceivable requirement can be exorbitant. Cost-effectiveness is of vital importance. Future operational requirements must be carefully developed and clearly defined so as to ensure the most economical solution. Future fighter aircraft must be designed with sufficient flexibility to meet the changing needs during their service life. Research and development should be directed towards those areas offering the most cost-effective solutions.

Recommendations arising from the Symposium may be divided into two groups for consideration, respectively, by the Military Committee and by AGARD itself.

To summarise the recommendations addressed to the Military Committee:–

(a) Operational requirements must be carefully developed in view of the ever increasing cost of fighter aircraft.

(b) Full use should be made of the available technology which should be challenged to produce new cost-effective solutions, particularly in the sub-systems area. However, the cost penalty of overspecifying aircraft and systems performance must be recognised.

(c) Future fighter aircraft designs must retain flexibility to meet changing requirements in their long life time.

(d) Requirements should recognise the use of the aircraft in its intended combat role and not over emphasise conflicting peace time needs.

(e) Requirements should clearly define the minimum quantity of weapons and external stores necessary to give the required military capability. Excessive capability will result in unnecessary adverse impacts on other characteristics.

(f) Consideration should be given to spending more in the development of a new aircraft to save still more in its Service life.

(g) The conflicting solutions offered by many relatively simple aircraft or fewer sophisticated aircraft still merit careful consideration.

For AGARD, its Panels and the Aerospace Applications Study Committee it is recommended that:–

(a) The lessons learnt from operational experience with current aircraft should be addressed so that they can be applied with benefit to future designs.

(b) Support should be given to:–
Developments in active control technology in its many aspects.
The use of piloted simulation for system studies, design development and training.
Further study of external store integration in fighter aircraft design.
Further research in inlet design, emphasising tolerance to high angles of attack and sideslip.
The use of wind tunnels, particularly in the development of specific configurations.

(c) The AASC should undertake studies to explore potential technology applications beyond those specifically requested by the Military Committee.

(d) On the structual side, support should be given to:–

Progress towards international standards for structural materials, particularly new materials such as composites.
Interchange of information on experience with composite structures in flight.
Development of new metal technology for cost effective applications.

(e) Further development of effective analysis techniques for use in establishing system and design requirements should be encouraged.

In conclusion, the Symposium was successful in addressing a broad spectrum of the many aspects of Fighter Aircraft Design. It brought together experts from the spheres of government, military and industry, exposing many of them to areas of technology outside their normal experience. While many aspects of design were not addressed and many questions were left unanswered, it is hoped that the Symposium will have served its purpose as a catalyst for many future activities and discussions in AGARD and the NATO community.

H.ANDREWS
R.J.BALMER
Members, Flight Mechanics Panel.

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** Paper not available. This was a summary of the AGARD Structures and Materials Panel Conference Proceedings Number 228.
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TECHNOLOGY DEVELOPMENT TO MEET THE MILITARY REQUIREMENTS

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1. INTRODUCTION

It is not the intention of this paper to discuss the means by which technology is developed. It is familiar to most of us and calls for some talented engineers, even a few scientists if available, an indication of the direction in which progress is required, and a sizeable part of the gross national product! Given this mix and the passage of time there is no problem in the development of technology. Indeed we are sometimes embarrassed with our already burgeoning technology – nowhere is this more true than in the field of avionic development where the future potential for the application of the ever widening range of sensors together with the associated data processing and control systems seems almost limitless.

Instead therefore attention will be drawn to some of the areas of technology which seem likely to earn, or to retain, their place in new fighter aircraft whose service life must now be expected to extend into the next century.

But why should a European, address you on this subject? Is it because European fighter development currently enjoys a World lead? Well hardly. We have seen Sweden plough a lonely furrow leading to the formidable Viggen, but this aircraft has yet to achieve an export success. Italy having built the Fiat G91 in the late 1950's and the G91Y more recently, has since largely equipped its Air Force with American fighters built under licence. The U.K. decided twenty years ago that the manned fighter was about to be replaced by the surface-to-air-missile, and has, after the Hunter and the Lightening, concentrated on the development of tactical ground attack aircraft. Only France, with the very successful line of delta winged Mirage fighters, followed by the later and conventional F1, has equipped her Air Force exclusively with Nationally produced fighter aircraft. All the other Western European nations have, in whole or in part, equipped their Air Forces with American fighter aircraft. With the appearance in the 1970's of the F14, F15, F16 and F18 there is little likelihood of an early change in this trend. So I can only explain my presence before you on two grounds, firstly it would be rather dull for the American members of the audience to travel to Europe just to talk to other Americans and, secondly, because I do not have something to sell you at present and this may allow me to be more objective in my outlook!

2. FIGHTER CLASSIFICATION

I propose to recognise three types of fighter aircraft as follows:–

(i) The Interceptor Fighter

Usually directed towards a non-manoeuvring target by ground control radar. Requiring high speed and high longitudinal acceleration it may also require high altitude and long range performance.

Emphasis on turning performance is not predominant and there is little need to consider low speed combat. Long range, all-altitude missile armament and associated radar is required.

The aerodynamic design is dominated by the need for both low profile drag and low wave drag.

(ii) Air Combat Fighter

Meeting its opponent by ground direction, by search of by chance. If an initial pass has been unsuccessful it may be assumed that the target has seen the attacker. Usually operating from the first line of bases behind the battle line its airfield performance is important and its primary operating zone is likely to be at low or medium altitude. The highest supersonic speeds are not necessarily required but the best possible sustained and instantaneous turning performance is essential. Short and medium range missiles as well as gun armament will be carried.

The aerodynamic design is dominated by the search for low induced drag at high 'g' and by the need for good departure characteristics as the dynamic stall is approached.

(iii) Ground Attack Fighter

Traditionally the first stage on the way to retirement of the interceptor or air-combat fighter but now differing increasingly in its requirements so that specialised ground attack aircraft are becoming more common.

Operating primarily at low altitude and with less emphasis on turning performance the wing loading can be much higher than for the other fighter types. Indeed a high wing loading may be desirable in improving the ride quality and the aiming accuracy achieved at the target in conditions of rough air.

There is no clear need for supersonic performance but the aerodynamic design will emphasise a high drag rise mach number and a store carriage arrangement which does not ruin the overall aerodynamic efficiency on the way to the target.

There is no significance in the order in which these three classes have been described. It would be better to think of them as occupying the corners of a triangle so that each is equally related to the other two. Along the sides of the triangle there is room for hybrids compromising the ideal characteristics of two of the three types. For example the Tornado might be regarded as lying along the side joining 'ground attack' and 'interceptor' while the F15 would lie more nearly on the line joining 'interceptor' and 'air combat'. Nevertheless there is no such thing as purity in engineering and in reality all fighter aircraft will occupy a position within the area of the triangle. I have been told for example, that the F15 was designed under the motto "not a pound for air-to-ground". The meaning being that no weight was intended to detract from the F15's Foxbat-catching ability in order to help it compete with, say, the A10. Nervertheless it is obvious that the F15 will still have a considerable ground attack capability.

So all fighters are to some degree multi-role even if, like the non-profit-making company, they did not set out to be that way.

The guy you should not believe is the one who will tell you that his product excells at all three corners. In reality he is somewhere within the triangle along with all his competitors.

I now want to go on to discuss airborne agility since this is the common factor which most-nearly characterises all fighter aircraft and is predominant in the case of the air combat fighter.

3. AIRBORNE AGILITY – The Search for a Parameter

If you want to beat the enemy's best fighter then you should be superior to him in all the agility-determining parameters. Obvious perhaps and probably expensive. So which parameters really matter? History tells us that the best fighters had high thrust/weight ratios and moderate wing and span loadings. Unfortunately it is not possible to optimise these parameters simultaneiously due to the interaction of wing weight.

With the advent of the computer and the simulation of air combat with experienced pilots 'fighting' each other in the simulation, an elaborate search has begun to determine the parameters which best characterise the most successful fighter aircraft.

A listing of all the parameters which have been proposed is not attempted. Sufficient to say that so far as the writer is aware no two groups have independently come up with the identical parameter. If we look for the commoner features then it is usual to find thrust, maximum usable lift coefficient, mean chord, span and aircraft weight. The first three of these tend to occur to an index of one or less, while the last two occur to an index of one or above. The term $\frac{b}{W}$ can sometimes be found raised to the power three. The implication of this last index to the structural design of the air combat fighter is obvious, the minimisation of the weight of a wing of the greatest possible span will place our structural colleagues in the forefront of effort on the next generation of combat fighters. All the parameters examined place emphasis on the aircraft turning performance, both sustained and instantaneous, although agreement is not unanimous on which is the more important.

As might be expected the progress of work on simulated combat has been extended from the simple one-on-one conflict to include one-on-two and two-on-two, and it is now being extended to include many-to-many in spite of the complexity (which was recently characterised as the mathematical equivalent of simulating a three-dimensional game of rugby). I question if this work will yield parameters more useful than those discussed above and I wonder if we have really progressed beyond the stage reached in describing a sports car by its engine capacity, its weight, the height of the centre of gravity and the wheel track. All appropriate no doubt but hardly sufficient to allow us to proceed straight to the design of a Porsche or a Farrari, or to distinguish adequately between the competing merits of theses famous products.

The simulators may serve a more useful purpose in 'screening' new inventions and in determining the value of short term dynamic effects, in order that the number of ideas that progress to the more expensive flight test stage is limited. Examples, most of which have already trodden this path include:- direct side force control, direct lift control, fuselage pointing both in yaw and pitch, variable sweep, vectored thrust, reverse thrust, sustained 'g' levels above the pilot's normal tolerance and differing missile and sighting systems.

3.1 The Acceleration Vector Envelope

If I am less than totally convinced by the 'magic number' approach, then what other methods have we for measuring the effect of design change on agility?

It has been common in recent years to plot specific excess power (SEP) against sustained turn-rate at given Mach number and altitude and to regard the area of the diagram between the positive axes as a measure of manoeuvrability. But this diagram does not do much to assist one's physical intuition. Since SEP is proportional to longitudinal acceleration and turn rate is (very nearly) proportional to normal acceleration, a far more useful diagram results if the two accelerations are plotted against each other. The resulting diagram is then the envelope within which the resultant acceleration vector of the aircraft can be directed. I am indepted to Alan Woodfield of MOD(PE) for drawing attention to this very simple conversion of the SEP/turn rate diagram and for illustrating its usefulness in comparing various design features.

Fig. 1 illustrates the simplest form of the diagram for an aircraft which at sea level and mach number of 0.4 has a maximum longitudinal acceleration in level flight of about 1g, is capable of sustaining about 6g (at which point available longitudinal acceleration is reduced to zero) and has structural design limits of +8 and −4g. The effect of closing the throttle and of extending the airbrake is also illustrated to complete the envelope within which the resultant acceleration vector can be directed while maintaining altitude.

Instead of plotting longitudinal and normal 'g' we have divided by the relative pressure and plot $\frac{n_x}{\delta}$ and $\frac{n_z}{\delta}$ so that the diagram becomes independent of altitude (apart from Reynolds number effects) and we can repeat the diagram at other mach numbers to cover the probable speed range of close air combat. One might suggest at this stage that the integration of the areas of the acceleration diagrams over the usable mach number range will yield a figure-of-merit for the agility of the aircraft under study which might be useful for comparative purposes.

The area of the acceleration envelope is not the whole story however since it is not instantaneously available to the pilot. If the pilot is flying in stabilised flight at (a) and then demands maximum longitudinal acceleration by opening the throttle and selecting reheat it will take a few seconds for the acceleration vector to extend to (b). If the pilot wishes to reach (c) he must roll the aircraft through substantially 90^{o} and increase the pitch attitude and this can be achieved within the engine acceleration time, if started from point (a) but will add a second or two if started from point (b). Point (d) can be reached within the time to roll through 90^{o} and increase pitch attitude as can point (e) since engine thrust decay is very fast. Points (f) to (g) can be reached in the time to extend the airbrake, say 2 to 5 seconds. If the pilot wants more than $-4g\ n_z$ then he must roll the aircraft through 180^{o} and pull through and this again may take 2 to 3 seconds.

If the pilot wants higher acceleration in the initial negative 'x' direction than can be provided by the airbrake then he must first turn through 90^{o} in azimuth and even at the high rates of turn now expected this will take perhaps 5 seconds depending on Mach number and altitude. To generalise then it may take between 2 to 5 seconds to reach the envelope of possible accelerations.

In the next generation therefore we must look for better engine acceleration times, fast operating airbrakes, very high rates of roll and pitch (the latter perhaps assisted by direct lift control) and with the ability to stabilise very rapidly at the demanded attitudes (and here the auto flight control system can help) together with the highest 'g' tolerance for structure and aircrew that we can obtain.

Returning to the acceleration envelope and means of increasing its area., Fig. 2 illustrates the effect of a 20% increase in thrust and the effect of a 20% increase in wing lift. Note that the negative n_z part of the diagrams has been omitted for simplicity. The increase in lift might be obtain by an increase in C_L or by an increase in wing area and the diagram would differ in shape a little depending on the precise drag and weight increases which resulted, but a single diagram is sufficient here to illustrate the qualitative effect.

The next diagram (Fig. 3(a)) illustrates the result of using a variable sweep wing. Assuming that the sweep angle is optimised to suit the normal acceleration demanded (reducing sweep with increasing 'g') then the result will be to reduce the longitudinal acceleration at low 'g' levels, due to the additional structural and mechanical weight of the wing, and to increase it at higher 'g' levels due to the greater available span and the resulting reduction in induced drag. The area of the acceleration envelope may well break even and if this is the case it will not assist in a judgement of the value of variable sweep. It is in just such circumstances, as mentioned earlier, that a recourse must be made to the differential manoeuvring simulator, or better still to flight experience. In short the acceleration envelope is not useful in judging the relative value of areas disposed in different parts of the diagram. Not all directions of the vector bear the same value in combat.

Finally Fig. 3(b) illustrates the effect of thrust vectoring. Once again there will be penalties in weight and thrust due to the installation of this feature but on the positive side note that engine thrust is available to increase the normal acceleration beyond the wing structural limit, although at its peak this will be associated with a negative longitudinal acceleration. (Note that the diagram is simplified in showing a circular 'peak', in reality flow interferences with jets deflected will distort this shape). On the negative longitudinal acceleration side the diagram is enormously extended, and these negative accelerations are very quickly available — really as fast as the pilot will move his nozzle control — so that the ability to force an adversary to 'fly through' is good.

Although not illustrated here it should be noted that as speed is reduced the vectored thrust diagram shows to increasing advantage relative to a purely wing borne aircraft. If an adversary persists in a decelerating manoeuvre, for example a horizontal 'scissors', against a vectored thrust aircraft he is bound to lose — no matter how good his wing — since ultimately he must lose control while the VT aircraft can manoeuvre on reaction controls and deflected thrust down to a few tens of knots.

To conclude by remarks on the acceleration vector diagram. It is very expensive to extend the diagram to the right because, once drag has been reduced as much as possible, this means more thrust and engines are expensive. It is cheaper to extend the diagram along the 'z' axis by means of more wing span, area or lift coefficient and cheapest of all to extend the diagram to the left. There is a distressing tendency to find that when you have pushed out one part of the envelope it will have contracted somewhere else, due to added weight or drag. The diagram does not help us to judge for example the value of variable sweep in reducing the roughness of the ride at high speed and low level, or the value of vectored thrust in producing a strong transient nose up pitch on nozzle deflection so that the aircraft performs the so called 'square turn' but it does give us a useful tool for comparing design features, and it will become more useful as experience is built up in flight and in the simulators.

4. REVIEW OF CURRENT AGILITY CHOICES

4.1 Raised Pilot's 'g' Tolerance

Fig. 4 serves to illustrate the most probable region for close air combat, that is the region of maximum turning performance. The diagram is drawn for a high thrust/weight ratio, moderate wing loading aircraft and it will be seen that if combat is limited to low level, by the risk of surface to air missile attack, then it will be possible to sustain 8g or above within a substantial part of the usable combat zone. To take advantage of this fact we must examine ways of increasing the pilot's 'g' tolerance.

Fig. 5 illustrates the postures that have been used in the past and concludes that the lower right hand diagram may be suitable for the future. The medical evidence suggests that the reclining position together with 'g' suit and pressure breathing should contribute an extra unit of 'g' to the pilot's tolerance and that his ability to perform tracking tasks is similarly improved.

4.2 Wing and Span Loading

Fig. 6 illustrates the wing loading and aspect ratio of a number of well known aircraft types. As expected those aircraft designed for ground attack have higher wing loadings than those designed for air combat, with the interceptors generally intermediate. Our ambition to lower the induced drag of the air combat fighter will be limited, in the case of a fixed wing design, by the roughness of ride that can be accepted under turbulent atmospheric conditions. If the avionics and hydraulics systems can be made to move the wing flaps or slats fast enough then perhaps a new freedom will become available in the choice of wing and span loadings. Otherwise it seems to me that a combination of say 60 lb/sq.ft. loading at combat weight, and aspect ratio of around 3.5 is likely to remain as a practical limit for fixed wing aircraft.

Fig. 7 illustrates, for the same aircraft types, the span loading parameter (W/b^2). The F14 illustrates the powerful effect of variable sweep – which no doubt also goes a long way to compensate for the high wing loading of this design seen in the last figure. The Mig 21 combines its moderate wing loading with the highest of the span loadings – a characteristic of the delta design which limits its usefulness as a close combat fighter. Again the Harrier and Jaguar exhibit span loadings appropriate to their primary role of ground attack. The F4 shows evidence of its original design as a long range naval interceptor. The latest combat fighters the F15, 16 and 18 exhibit span loadings about half that of the earlier generation Mig 21.

You will remember that the combat parameter approach called for the highest usable lift coefficient so that in association with a moderate wing and span loading we shall expect to use full span leading and trailing edge flaps at optimum deflections throughout the combat range. If we have optimised the wing camber for the subsonic hard-turning combat then the leading edge flaps may be used to reduce this camber for transonic and supersonic operations.

4.3 Engine Characteristics

In full reheat, engine thrust is determined (very nearly) by the air mass flow at maximum achievable RPM and the exhaust temperature. The particular thermodynamic cycle of the gas producer providing the mass flow is relatively unimportant. This tells us that the engine should have the highest possible flow density and the highest possible reheat temperature. But what of the dry thrust? Fig. 8 illustrates for a number of fighter aircraft the dry and reheated thrusts at M = 0.8 sea level. We should remember that at maximum dry thrust the fuel consumption may already be twice that at the clean cruise and for the later designs the fuel flow at maximum reheat may be multiplied by a further factor of 5 or 6.

So with an overall factor of about twelve, an aircraft that could cruise for an hour at low altitude would exhaust its fuel supply in full reheat in only five minutes. While we may hope at least to double that time it remains a fact that the combat time in full reheat is measured in minutes. Now this is very unfortunate because it is quite clear that to break-away and run from close combat is now extremely hazardous. With turn rates of up to 25°/sec. available, it follows that even if the last pass was head-on, then the combatant who is able to remain on the offensive can send a missile after his departing opponent within perhaps 10 seconds. This raises the concept of 'persistence in combat' and emphasises the need for an adequate dry T/W ratio to limit the excurtions into reheat.

Furthermore at a given mass flow (for instance at maximum RPM) the exhaust temperature and associated metal temperatures are roughly proportional to fuel flow and at its simplest the IR signature is proportional to the fourth power of the temperature. I did say I had nothing to sell but note the Harrier which, although subsonic, can, at low altitude, force many of its adversaries into reheat and then may win the waiting game that follows.

Overall therefore we shall look for engine cycles with only moderate by-pass ratios – too high and the fighter will spend too long in reheat for the good of its health; too low and its cruise efficiency will suffer and the fuel fraction will once again be driven up, resulting generally in a larger vehicle.

5. SURVIVABILITY

The previous sections have sought an aircraft which by its agility can expect to bring its weapons to bear on an airborne adversary – and to avoid the reverse situation. But no amount of agility will protect against the unexpected attack either from the air or the ground. So what feature will minimise this risk?

The signals which can betray an aircraft's presence are visual, radar and infra-red (sound may still draw the attention of those on the ground but is not a very suitable signal for a homing missile) and all three can be reduced by keeping the aircraft small. Fig. 9 shows three pairs of fighter aircraft. In each case one is large and the other small. It is not suggested that each aircraft in a pair was designed to the same requirement – e.g. the Hunter has twice the gun armament of the Gnat and the Phantom is a two-seater and has a larger radar and fuel fraction than the Mig 21 – but rather that aircraft designed to quite different requirements may end up in conflict, as in the first and third examples. (In the middles example both types still serve with the Indian airforce and after the Indo-Pakistan war the Indian Government decided to put an improved version of the Gnat back into production). Once again small is beautiful and probably cheaper too. Three elements can assist in achieving smallness (i) advances in technology tending to miniaturisation (ii) skill in design leading to dense packaging and (iii) restraint by the military staff writing the requirement, in order that the 'kitchen-sink' is not taken along for the ride!

The fuel fraction is an important contributer to size so that the long range fighter will always be at a disadvantage against a local defender.

5.1 Reducing the Signal Strengths

Given the smallest overall size, other methods of reducing the signal strength include:–

Visual signal – A smoke free exhaust is essential with or without reheat and this should be a mandatory requirement for any new fighter engine.

We must pay more attention to camouflage. Useful work has been done in the last few years, mostly in the United States. We need to change the colouring with the seasons and the geographical location. (In Europe – almost from day-to-day depending on the weather!).

Radar Signai – We should avoid surfaces at right-angles to each other to limit the number of corner reflectors. This may be dominated by the wing/fuselage junction, particularly so if a high wing is used, and this suggests that the root chord should be small and that wing leading edge strakes may be adverse in this respect. One may also wonder if the rectangular intake ducts of recent years are a good feature from this point of view.

The intake duct itself may be treated with radar absorbent material and the throat area should be minimised by the use of blow-in doors or hinging lips as part of the aerodynamic design. If the intakes could be placed above the airframe then they would receive some screening from ground radars, however the aeodynamic problems of achieving satisfactory intake performance at high incidence are obvious.

The cockpit transparencies may be coated with metallic film to reduce the large radar returns otherwise common from within the cabin.

We need to know more about the radar reflective characteristics of carbon fibre composite aircraft structures, and whether or not these may need to be metal sheathed anyway to protect internal systems from external electro-magnetic radiation.

IR Signal – While the detector heads fitted to surface-to-air and air-to-air missiles are becoming increasingly sensitive so that boundary layer heating can make the whole airframe a significant signal source, the problem is still dominated by the engine. As already noted the high thrust weight ratios now demanded can be provided only with the use of reheat and the resulting IR signal is enormous. Furthermore the development of very compact reheat pipes with short burning lengths has increased the angular range within which the strongest signals are transmitted.

It may sometimes be possible to provide a degree of airframe screening e.g. the tailplane of the F4 or Jaguar, external stores on the Harrier, or the aerodynamic nozzle of the Viggen (Figs. 10, 11, 12) but it seems that we shall depend heavily on the development of decoy systems for survival against IR homing missiles.

Finally, and as mentioned earlier, the dry thrust should not be at so low a level that the aircraft is forced frequently to fly in reheat with the resulting greatly increased IR signal strength.

5.2 **Survival in Spite of Battle Damage**

We have looked for an agile aircraft and one that advertises its presence as little as possible. What steps can be taken to improve its chance of survival if in spite of this it does receive battle damage?

The three most vulnerable areas are the pilot, the powerplant and the fuel system.

We cannot make the pilot smaller, we can surround him with armour – although this implies a serious weight penalty; rather we should pack avionics equipment ahead or behind the cockpit as a means of stopping splinters. 'Flack curtains' or light armour may be justified on the cockpit floor and side walls.

With the emphasis on high powerplant thrust/weight ratio we can not assume that the engine casings will resist heavy fragments, still less direct hits, neither can we assume that a damaged engine will contain its debris. Leaks of flame or hot gas able to create fire damage are probable rather than possible results of engine damage.

If two engines are used then they should be sufficiently separated (as for example in the F14) so that explosion or fire in one does not rapidly lead to loss of the other.

The engine and its intake duct should be remote from the fuel tankage – difficult as this is to arrange. The engine nozzle(s) should receive whatever screening is possible from the airframe, and they should be as remote as possible from systems, tail surfaces and primary structure so that the result of IR warhead detonation in the vicinity of the nozzle may be limited to reheat pipe and secondary structure damage.

The fuel tanks should be integral, with as few tank surfaces exposed within the airframe as possible. The tanks should be foam filled and be unpressurised during combat. The highest threat tanks should be emptied first.

If an entirely satisfactory non-inflammable hydraulic fluid can not be found then hydraulic power should perhaps be limited to the flying controls and other services should be operated electrically or pneumatically instead – since these forms of power carry a lower fire risk.

Neither fuel not hydraulic fluid should be stored adjacent to, nor carried past, the cockpit area.

Following battle damage the aircraft must be repaired and if future wars are of the six day variety then we are looking for almost instant repairs to both structure and systems. It should be remembered that the ground crew may be constrained to wear NBC protective clothing during this process.

Finally we have discussed survivability in the air but this is of little value if the aircraft can not also survive and continue to operate on the ground. This brings in the question of shelters, operation from grass, and dispersed basing – all subjects receiving renewed study.

5.3 Concluding Remarks

I have not touched on armament or avionics except to note that these are areas of most active and even bewildering advance. We badly need an agile forward hemisphere missile. We must reduce the pilot's workload. We can not afford to carry a ton of avionics.

I hope that I have touched on enough topics to suggest some of the developments that are desirable.

If you can take all my good advice and synthesise it into a fighter aircraft design which still retains the simplicity of an army truck, then I believe the military will be very grateful to you!

The views expressed are my own and do not necessarily reflect those of British Aerospace. I thank you for your attention, and I look forward to learning from you during the remaining period of this symposium.

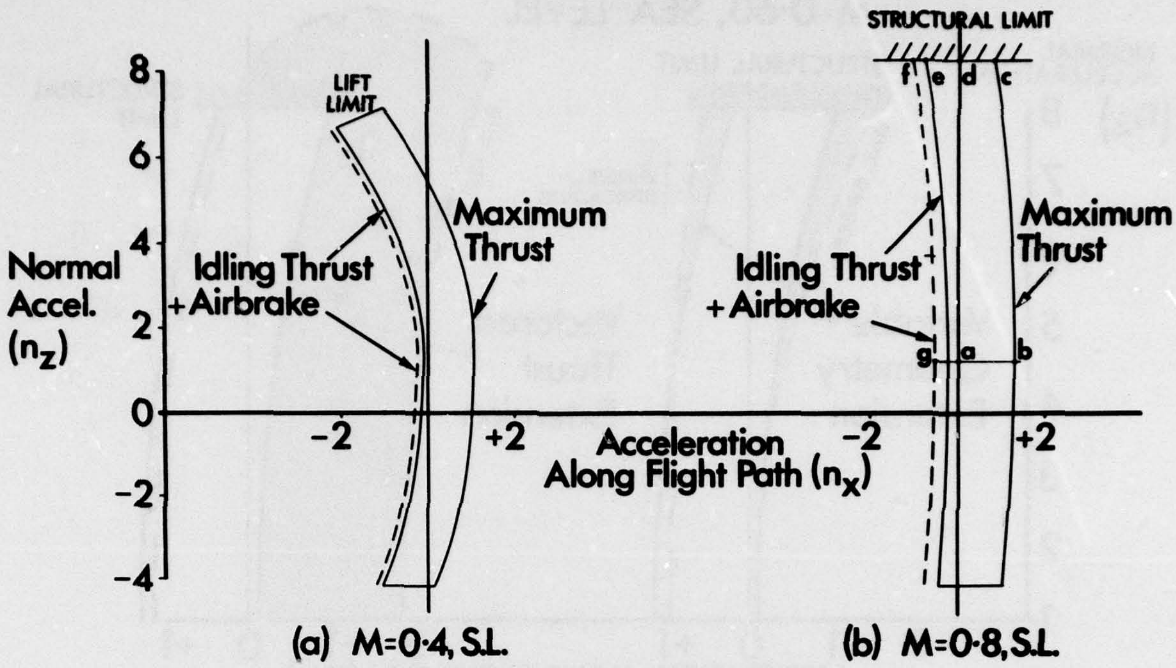

Fig.1 Acceleration vector envelopes

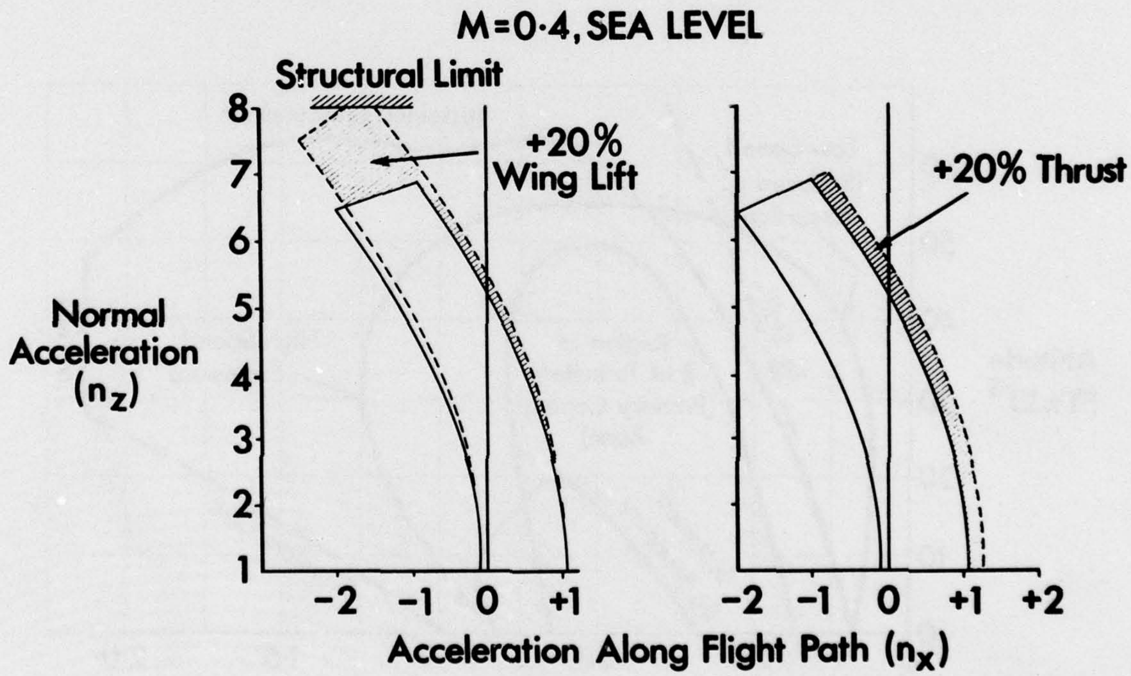

Fig.2 Acceleration vector envelopes – (Extra lift and thrust)

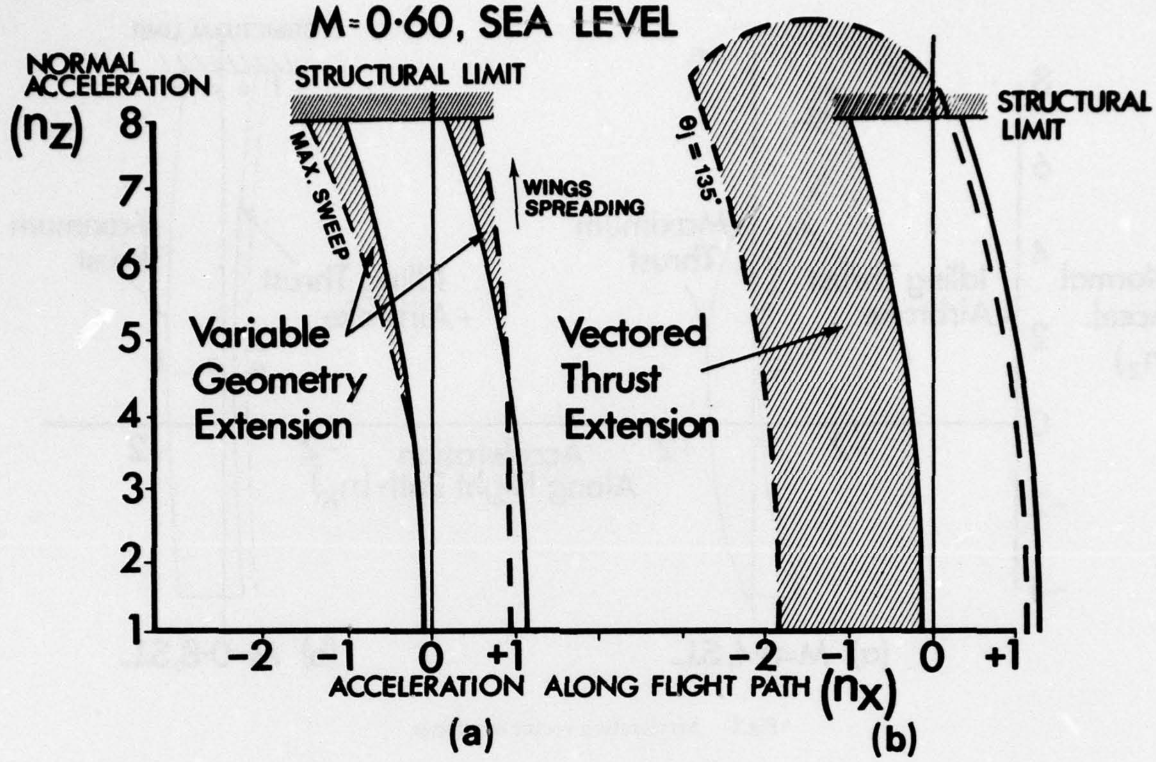

Fig.3 Acceleration vector envelopes – variable geometry and vectored thrust

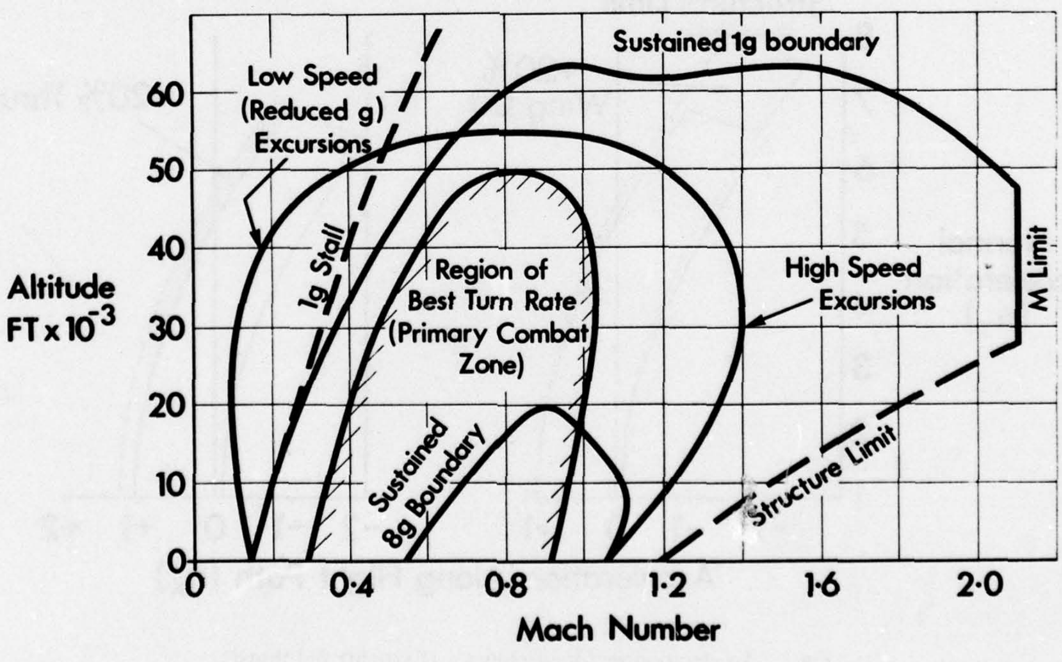

Fig.4 Combat zones

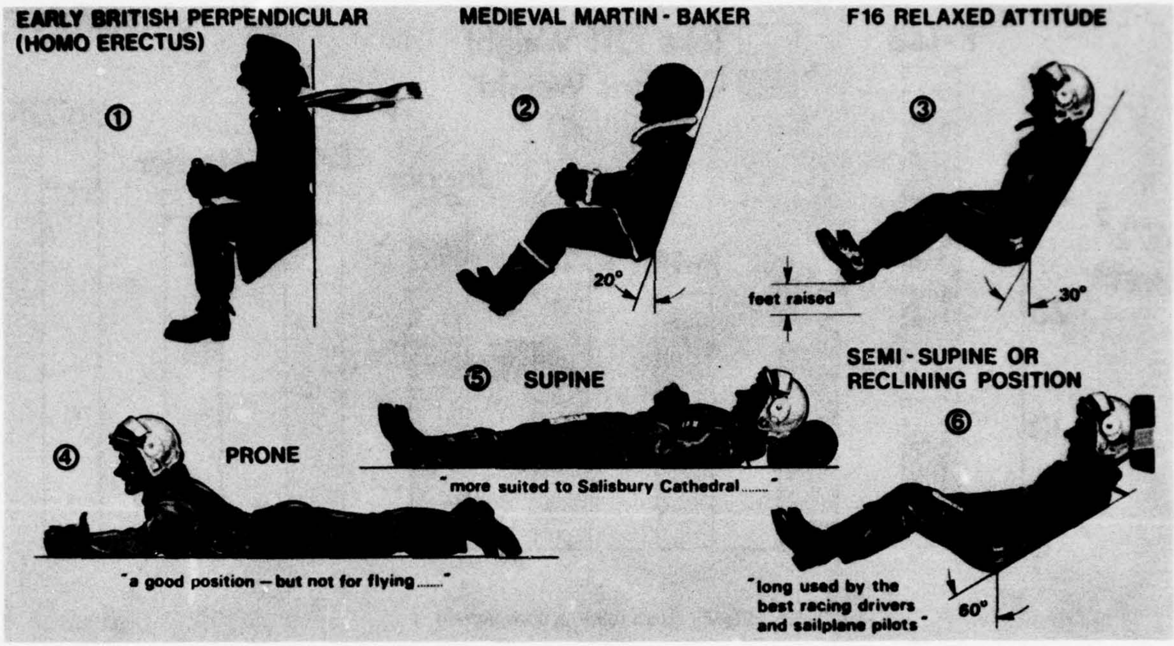

Fig.5 Pilot's seating postures

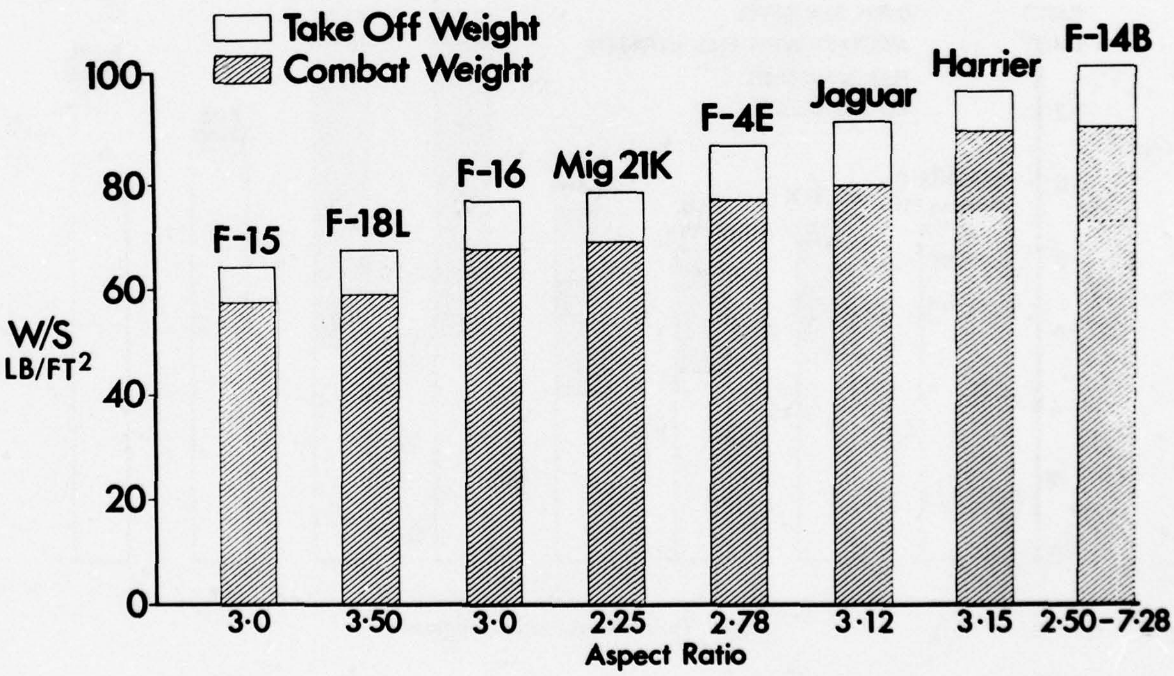

Fig.6 Wing loading comparison

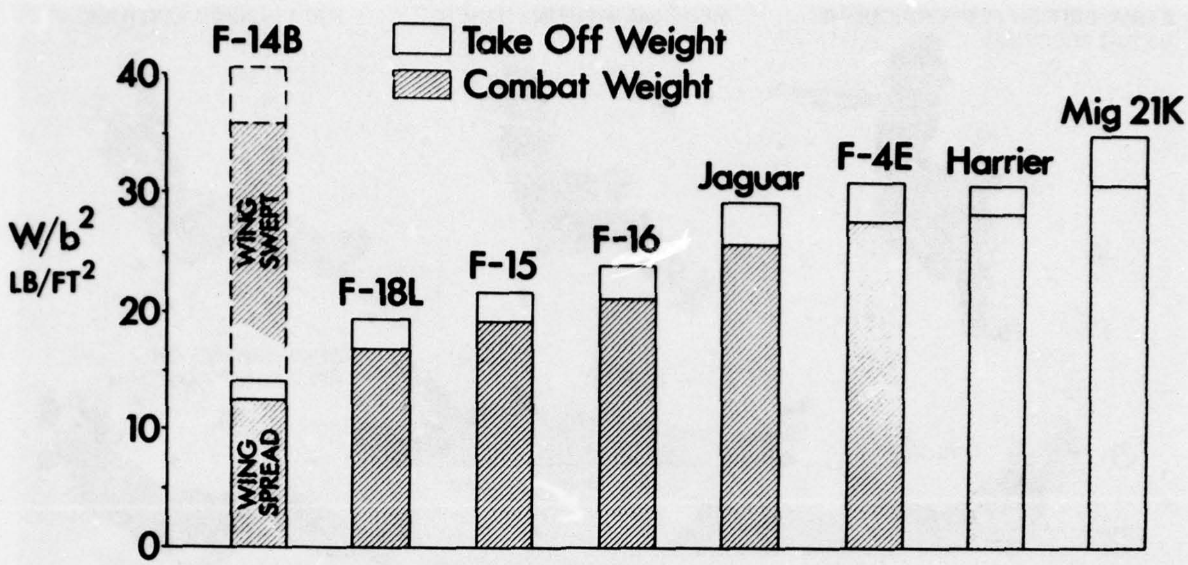

Fig.7 Span loading comparison

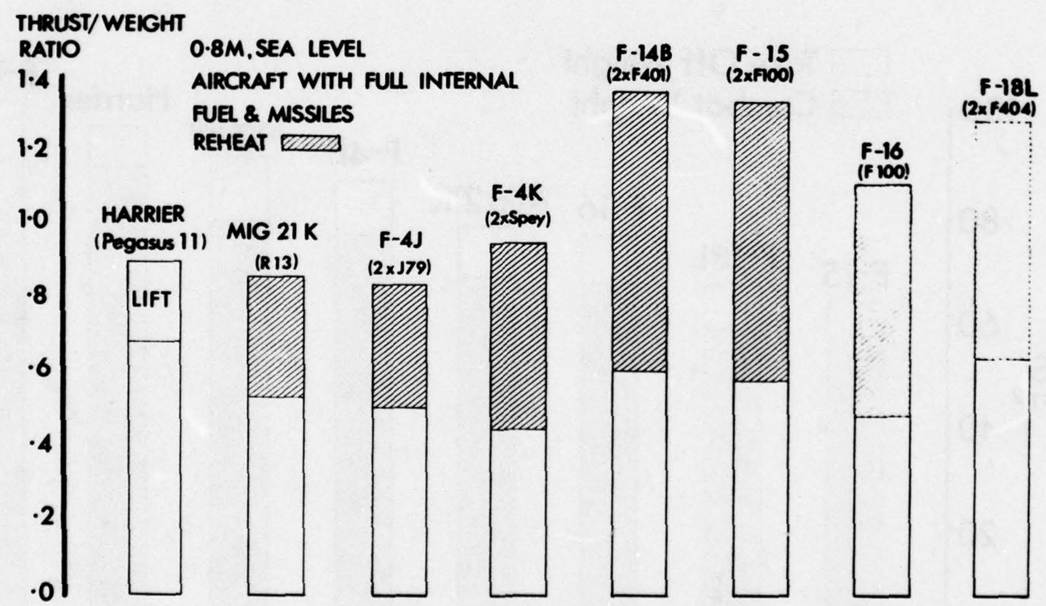

Fig.8 Thrust/weight ratio comparison

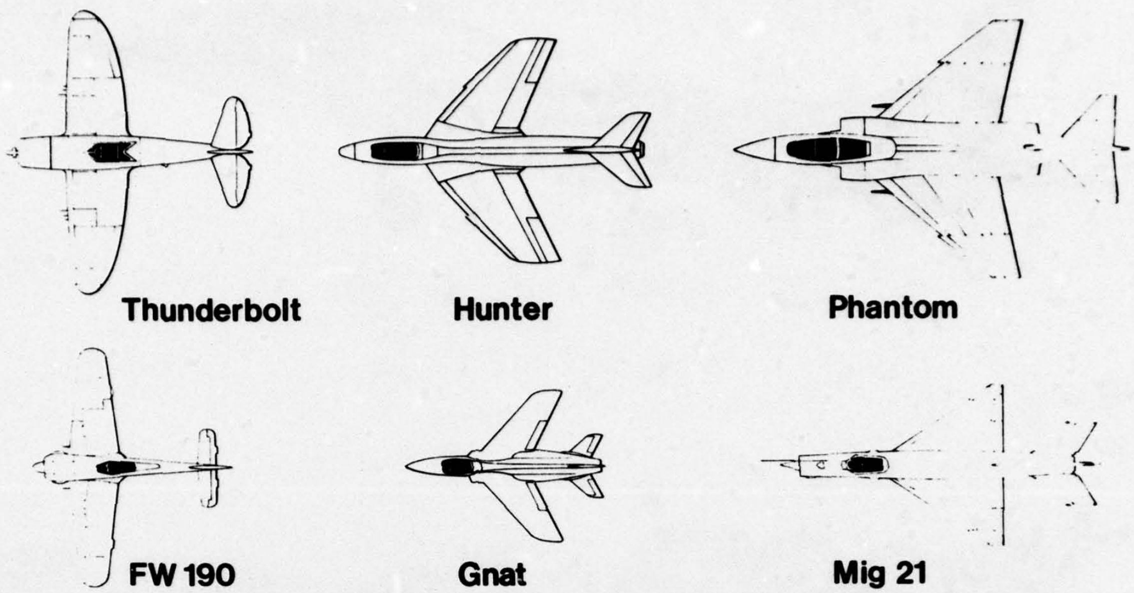

Fig.9 Fighter comparison

Fig.10 Jaguar with reheat on

Fig.11 Harrier TMk2

Fig.12 Viggen with reheat on

ASSURING COMBAT PILOT EFFECTIVENESS

by

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SUMMARY:

The problem of assuring combat pilot effectiveness is one that has continually concerned the Air Force. We historically have entered into combat situations at relatively low proficiency levels, and worked, or fought our way to higher levels of proficiency, attaining a maximum level of proficiency as the conflict ends. As we re-entered the peacetime environment, our proficiency levels go down. In an attempt to maintain a capability that will allow us to have a "mission ready" status on the first day of conflict, we have in the last four years, restructured our aircrew selection and training programs. Four relatively new selection and training concepts, Fighter Lead-In Training, the Graduated Combat Capability Concept, the Red Flag Program, and the Mission Simulator are discussed in detail. The Fighter Lead-In Program provides basic tactical air-to-air and air-to-ground training in a high performance trainer aircraft. This provides a better basic tactics training, and additional crew selection at a cost that is lower than operating tactical aircraft. The Graduated Combat Capability concept provides a means of assigning and maintaining a series of specific mission capabilities to a unit. The concept provides for high proficiency levels to be maintained, and capabilities added or deleted based on unit manning, aircraft availability, etc. The Red Flag Program provides a complete mission scenario in which the performance of a particular crew member and the entire unit is carefully monitored and assessed. The Mission Simulator concept provides for high fidelity training in areas that are expensive, impractical, or potentially unsafe in the actual aircraft. A takeoff to touch down capability is provided, with the ability to repeatedly train in specific tasks within the mission. The impact of this specialized training philosophy on the design criteria for specialized airframe and availability is also discussed.

INTRODUCTION: A LOOK AT FIGHTER AIRCREW TRAINING HISTORY:

Each aerial conflict has brought one very concerning fact to light. As we enter into combat, our initial proficiency levels begin at a relatively low side, and then build to higher levels as the conflict is resolved. A graphical portrait of this is shown in Figure 1. This can be attributed to the less proficient personnel becoming victims of the more skilled, and to personnel who do survive becoming more proficient. An example of this is a statement made by Major Richard Bong, the leading American Ace in World War II, with 40 victories in the P38 Lightning. Major Bong flew two combat tours; his second as a gunnery instructor. Major Bong stated that if he were as proficient in aerial gunnery on his first tour as he was on his second, he would have easily had over 80 kills to his credit. Figure 1 shows the two conflict phases that took place during the North Vietnam conflict, referred to as Linebacker I and II. As we return to a peacetime posture, we lose these basic skills of combat proficiency at a much greater rate than we acquired them. We assume that our skills, or proficiency levels have increased as a result of each conflict, as portrayed by the increase in average proficiency with the passing of time. Our current objective is to examine what factors in the various mission tasks have resulted in our successes (and our failures) and determine how our peacetime training programs must be restructured to maintain proficiency levels so that each crew member is fully capable of successfully completing his first mission on the first day of conflict.

As technology progresses, the inherent capabilities of fighter aircraft have expanded to fill a variety of roles. We see current inventory aircraft, such as the F-4 Phantom II being used as an interceptor, air superiority fighter, for deep strike, fighter close air support, and reconnaissance, while possessing the ability to defend itself against hostile ground and airborne threats. In order to gain insight as to how we have transitioned from the classical training program with its known shortfalls to the program we have today, we will present a brief look at our history in Vietnam with a current aircraft.

Throughout the period of the Vietnam conflict, tactical aircrew training changed very little. The only additions were to accommodate the new weaponry that had evolved during the conflict, and in each case a special training program was developed. This, however, did not change the basic philosophy of training or the content of the existing programs. The backbone of this philosophy was contained in Air Force Regulation 60-1. The training structure it provided was based on the minimum number of hours and events (such as instrument landings), which a pilot was required to complete in each six month training period. This regulation applied to all pilots in the Air Force and provided for the basic training necessary to assure safety of flight. It did not provide aircrews with the training necessary to successfully employ an aircraft in combat. This aspect was left to another document; the 51 series manuals, one which was written to provide a training program for each aircraft. These manuals defined combat training lists of the minimum events which pilots would achieve in a training period. In some events, a qualification was required; weapons events such as dive bomb, were in this category. In addition, the Tactical Air Forces (TAF) planned and allocated 20 hours flying time per month for each squadron pilot. This being the historical measure of what was required to maintain his combat ready status. To ascertain if a pilot was maintaining his combat ready status he was given a tactical evaluation flight with a flight examiner, commonly referred to as a "TAC CHECK". If an aircrew had a current Tactical Air Command (TAC) check, was weapons qualified, and had completed the training events in the prior training period, then he was reflected as a trained aircrew in the units force status report, thus closing the management loop back to higher headquarters. To assure the validity of the system, the units were periodically given Operational Readiness Inspections (ORI's) to validate the force status report.

However, 1973 was a turning point for TAC. Operations in the previous year in Southeast Asia, particularly Linebacker I and II, pointed out the need for better air-to-air training. At the same time, with the reduction of the Asian conflict, the need for our forces in Europe and possible problems in the Middle East drew attention to the threat we faced there. At the same time, budgetary restrictions covering the last two quarters of FY 73 had forced a reduction in flying hours for primary aircrews in TAC from 20 to 18 hours per month. In May and June, the energy crisis was beginning to make itself felt.

"The forseeable outlook for the military services is significantly reduced availability of all petroleum products at increased prices."

General Horace M. Wade
Vice Chief of Staff, USAF, 2 June 1973

This was quickly followed by the Air Force Chief of Staff directives to TAC requiring reductions in operational and maintenance resources, and the use of fuel in 1974. At this point, the goal could be met by 18 flying hours per month for primary crews with a reduction in aerial flybys and static displays.

The second event to impact the problem occurred in October 1973 with the Arab-Israel Yom Kippur war and the oil embargo which resulted. TAC, with the rest of the Air Force, found itself so resources limited that it was necessary to take a new approach to training.

In developing a new training concept, there were lessons learned from the past that had to be considered. The reductions in resources, notably fuel, had shown that a program defined in hours and events may train aircrews but made it very difficult to determine what was really required for an aircrew to be ready for combat. The reason was that the aircrews had regularly exceeded the minimum, thus their performance in combat was the result of better (and more) training than that required. To confuse the issue further, it was very hard to determine how much had actually been accomplished above the minimum since only accomplishment of the minimum was reported. This made it very difficult to justify the need for flying hours programs which would provide more event training than was required. This problem was reinforced by not being able to express exactly when an aircrew would lose combat capability or would become unsafe in flying a particular aircraft. Safety studies helped but findings were general in nature when specific answers were being sought. Southeast Asia had left doubts as to the adequacy of the air-to-air portion of the training program.

The 1960's saw development of fighter aircraft capable of performing many missions, air-to-air, conventional and nuclear ordnance delivery. In addition, new weapons such as laser and TV guided bombs and missiles were coming on the scene. Not only was the complexity of the aircraft and mission increasing, but the defense, with large numbers of surface-to-air missiles were requiring more attention in training in order to avoid them and still accomplish the mission. By 1973, the question was how much could an aircrew be trained to do and still have him at a level he could be instantly put into combat. Twenty hours a month wasn't enough. TAC's air-to-air program had consisted of six air combat training sorties. Our Southeast Asia experience contrasted with the success of the Israeli Air Force in the Yom Kippur war indicated a need to rebuild the air-to-air program (the result would be to increase the air combat sorties to fifteen).

The lessons learned from the Vietnam conflict, along with increased costs of fuel have resulted in a restructuring of fighter training programs. The training of today's fighter pilot begins upon graduation from undergraduate pilot training. The Tactical Air Command Pilot enters the Fighter Lead-In Training (FLIT) program upon graduation from undergraduate pilot training. Upon successful completion of the FLIT program, he then goes on to a Replacement Training Unit (RTU) to get his initial checkout in the aircraft that he will fly. Upon graduation from the RTU, he will then be assigned to an operational unit where he will then begin his tactical qualification check, local checkout, theater checkout, and finally achieve mission ready status under the Designated Operational Capability (DOC) concept. Once awarded mission ready status, the pilot then goes through a continuing process of maintaining proficiency at the various tasks in his particular operational capability requirements. Two steps recently undertaken to help maintain mission ready proficiency are the Red Flag Exercise Program and the Mission Simulator. The remainder of the paper will deal in the four areas of the Fighter Lead-In/Expanded Fighter Lead-In Training Program, the Designated Operational Capability Concept, the Red Flag Program, and lastly, the Mission Simulator Concept. The chronology for training a fighter pilot is shown in Figure 2, which plots the various courses, or programs that the crew member must successfully complete prior to achieving mission ready status and the various programs that he must participate in to maintain mission ready status in an operational aircraft.

THE FIGHTER LEAD-IN PROGRAM:

The Fighter Lead-In Flying Program using the T-38 aircraft was started in early 1973. The purpose of the program was to train students who had recently completed undergraduate pilot and undergraduate navigator training in basic fighter tactics with an aircraft that they were familiar with, the T-38 Talon trainer. These aircraft are being modified to carry a practice bomb dispenser and mini-gun pod. They are being fitted with an optical fixed depressed reticle gunsight. The program called for each student pilot to complete a total of 22 missions of which the composition is shown in Table 1.

TABLE 1

FIGHTER LEAD-IN FLYING TRAINING

	PILOT	WEAPONS SYS OPER
TRANSITION	1	0
FORMATION	7	2
BASIC FLIGHT MANEUVERS	8	4
LOW LEVEL NAVIGATION	1	1
GROUND ATTACK	5	4
	22	11

Each weapon system operator entering the Tactical Air Command was to complete a total of 11 sorties in the F-38 aircraft to familiarize him with the Tactical Air Force mission. In addition to the flying and the training portion of the Fighter Lead-In Program, 60 to 100 hours of ground or academic training was also included in this particular program. The particular sub-areas of academic training are listed for the Fighter Lead-In Program in Table 2.

TABLE 2

FIGHTER LEAD-IN ACADEMIC TRAINING-HOURS

	PILOT	WEAPONS SYS OPER
SPECIALIZED	7	6
LIFE SUPPORT	3	3
FLIGHT CHARACTERISTICS	0	4
AIRCRAFT SYSTEMS	4	4
BASIC FLIGHT INSTRUMENTS	0	10
BASIC FLIGHT MANEUVERS	17	17
FORMATION	4	4
MISSION PLANNING	3	3
WEAPONS DELIVERY	4	4
AIR ATTACKS	0	4
RADAR AND INS	0	16
ACADEMIC PREPARATION	21	27
TOTAL	63	102

After approximately one year of operation the benefits of this program became fairly obvious. For students entering the F-4 aircraft, approximately four missions could be eliminated from the F-4 program which resulted in net savings of approximately $1200 per student. For those students entering the A-7 aircraft, four sorties could also be eliminated from that program for a net savings of approximately $1,000 per student. In addition to these cost savings, additional proficiencies were gained with a familiar aircraft in the fighter tactics mission role. The Fighter Lead-In Program was judged to be so successful that a new program was initiated in early 1977. This program was known as the Expanded Fighter Lead-In Program (EFLIT) and increased the number of sorties that each UPT student flew from 18 to 49 missions in total. Table 3 shows the comparison between the basic Fighter Lead-In Program and the Expanded Fighter Lead-In Program in terms of the increased number of sorties that each student was required to successfully complete prior to entering the operational aircraft that he was designated to fly.

TABLE 3

EXPANDED FIGHTER LEAD-IN FLYING TRAINING

	PILOT
TRANSITION	2
FORMATION	7
BASIC FLIGHT MANEUVERS	17
AERIAL COMBAT MANEUVERS	4
LOW LEVEL NAVIGATION	3
GROUND ATTACK	13
GROUND ATTACK (TACTICAL)	3
	49

As can be seen, the greatest number of sortie increases was in basic flight maneuvers; from 8 to 17 sorties, and from the ground attack; from 2 to 13 sorties. Tactical ground attack was included for an additional three sorties to make 49 sorties in the EFLIT program. The costs benefits that are accrued as a result of the Expanded Fighter Lead-In Program are that for students entering the F-4 aircraft, eighteen sorties can be eliminated from the replacement training unit program. For those students entering the F-15 aircraft approximately ten sorties can be eliminated from the Replacement Training Unit Program. Some additional benefits that have been accrued as a result of the Expanded Fighter Lead-In Training Program are that we are now able to more carefully screen and control those students who are performing marginally in terms of their ability to successfully complete the mission that the aircraft

they are training for is designated to complete. This screening and validation process gives us an opportunity to examine the students progress at two check points within the Expanded Fighter Lead-In Training Program. If the student's performance is marginal at either one of these two check points he is given the opportunity to complete a flying evaluation board or to request a waiver to fly in a less demanding aircraft or system. Our progress to date in this program has indicated that we have eliminated approximately three percent of the students who enter the Expanded Fighter Lead-In Training Program.

THE DESIGNED OPERATIONAL CAPABILITY (DOC) AND GRADUATED COMBAT CAPABILITY (GCC) CONCEPT:

In an attempt to maintain high level proficiency in a peacetime environment with limited resources, a new concept of training was initiated to address the problems that these resource constraints impose. This concept hinged about developing separate training programs for each of the basic roles or missions of a fighter aircraft. These roles were defined as air superiority, air defense, air-to-ground nuclear and air-to-ground conventional. This concept was known as the Designed Operational Capability (DOC) concept. To match the DOC system, a new reporting system, Unit Capability Measuring System (UCMS) was designed that required the unit to report its capability to perform the role or mission that it was assigned. This system also addressed the shortage of resources problem by defining three levels of proficiency by which pilots could be tasked to maintain. Pilots trained to a Mission Ready (MR) status in the mission were ready for combat. Those trained to a mission capable status were less qualified and those at a basic proficiency level were not trained for combat but were capable of safely operating the aircraft. Manuals were written which defined in terms of mission that was required in each tactical aircraft to be trained at a given status in the DOC. These tactical aircraft to be trained at a given status in the DOC. These determinations were made on the basis of past experience of tactical commands and a realistic continuation training study conducted by the Tactical Fighter Weapons Center at Nellis AFB, NV. Additional flexibility was provided to the commander by including a regression point in each DOC program. This minimum allowed for mission requirement differences due to various degrees of experience between pilots. It provided that the new inexperienced pilots could be allocated more missions, while highly experienced pilots was given less missions resulting in maximum efficiency to be obtained. Units were assigned DOC's depending on the type of aircraft they possessed and the capabilities that were desired. Multi-role aircraft were assigned up to three DOC's, two at the MR proficiency level and one at the MC level. The DOC system was implemented in mid-July 1975 with the F-4 and other tactical fighter systems were soon to follow. The DOC system worked very well but with the introduction of new aircraft to the inventory such as the F-15 and new weapons to existing weapons systems such as laser guided weaponry, etc., it soon became evident that a more specialized breakdown within the DOC system was required. The answer to this problem lay in graduating the training program based on the degree of difficulty with each particular sortie within the four blocks of the DOC. By arranging sorties in a loose structure starting with less demanding tasks and progressing to the most difficult, it was possible to safely fly aircrews having very limited combat capabilities at levels of proficiency not previously associated with any DOC system. To cope with this, a new system of a graduated DOC, referred to as Graduated Combat Capability (GCC) was introduced. This concept is an evolution of the DOC concept which took into consideration the change and developments which caused problems in the basic DOC systems. The concept was designed as a continuum, starting with the threat to the requirements to counter the threat and resulted in an assignment of specific mission tasks to the unit and their training required to meet their intended mission. This concept took the DOC one step further by defining in each DOC the combat capabilities and specific training requirements defined in sorties to obtain an MR status in each capability to counter each threat. Each new weapon, such as Maverick was treated as an individual capability. Each capability was further quantified by providing two sortie levels, the sorties required for experienced crews and a more extensive program required for inexperienced crews. Figure 3 portrays how this particular system works. For a given requirement, a specific number of sorties are required to be flown. The more sorties that are available for a particular unit the more capabilities this particular unit can possess. The basic capability for an air-to-surface wing starts out with collateral support and then begins to pick up on nuclear strike, close air support then going to interdiction and finally guided weapons. Within the air-to-ground missions are the number of sorties required to maintain proficiencies with various weapons such as Maverick, etc. These can be further shown within that capability. The contents for each capability was developed by breaking the training needs to accomplish the capability down to its three basic elements. They are (1) those being the sorties required to accurately deliver the weapons, (2) the training necessary to get the weapons to the target, and (3) the training required to negate the defenses, aircraft, surface-to-air missiles, electronic countermeasures, and anti-aircraft artillery that would be encountered on the ingress and egress to the target. When tasking the units, the combat capabilities are assigned in order of priority. This provides an automatic management feature to the system which prevents dilution of higher priority training. Thus, as sorties are reduced for whatever reason, the unit drops from its training program the capability it can no longer support (the least priority).

OPERATION RED FLAG:

The Red Flag Program is a continuous training program to provide world-wide tactical fighter forces with a realistic enemy threat in the training environment on a continuing basis. The method of operation for the Red Flag Program uses a Red Force stationed permanently at Nellis AFB, NV., and a Blue Force is sent each month to fight the Red Force. The outcome of the battle is monitored with the White Force. The Red Force is composed of two components, the air threats and surface threats. The air threats are composed of F-5 aircraft from the aggressor squadrons which are based at Nellis AFB. Radar controllers, airborne electronic warfare, and communications jamming aircraft are also used in the air threat. The surface threats are composed of simulated enemy type anti-aircraft artillery and surface-to-air missile threats. The White Force is the key to success to the Red Flag Program. The White Force uses a large array of sophisticated data retrieval systems that assist in mission analysis and debriefing. The Blue Force is composed of line fighter squadrons from the Tactical Air Command, National Guard, and Air Reserve forces. These squadrons use F-4, F-15, A-7, and F-105 aircraft. In addition, support elements, consisting of reconnaissance aircraft, forward air controllers, electronic countermeasures, and search and rescue aircraft are used to supplement the Blue Forces. All the major air commands participate regularly, or have participated in Red Flag as Blue Forces. These include the Strategic Air Command with their B-52 and KC-135 aircraft, Air Defense Command with their F-106 interceptor aircraft, and the Military Airlift

Command in simulated aerial drops. Navy, Army, and Marine Corp aircraft are also utilized such as the A-6 and EA-6 electronic countermeasures aircraft and the Marine Corp A-4 attack aircraft, and Army helicopters. The Blue Forces deploy from their home base for a one month tour at Nellis AFB. The Blue Forces bring their own maintenance crews with them for the month and rotate two aircrews through the training program every month, this way we obtain two units worth of training for one units worth of mobility costs for every exercise program. The facility at Nellis AFB was selected to take advantage of the large areas of land and air space available. The first few missions at Red Flag are tailored to meet the needs and experience of the individual crew member and provide a base for quantification. Once the crew member is oriented as to where the ranges and facilities are, then the actual training program begins. Training takes place in those missions of interdiction, close air support, and air superiority. To evaluate the Graduated Combat Capability of a particular unit, the F-15 is used as an example. Training with this aircraft has developed into a wave attack scenario that provides F-15 pilots with a numerically superior force to fight. Their problem is one of prioritizing targets while operating their own radars against countermeasures and jamming provided by Navy EA-6 aircraft. F-4 aircrews are given training problems requiring the use of such weapons as the Maverick missile in a simulated high threat environment. This could possibly be used in conjunction with a surface-to-air rescue mission where the F-4 pilot might be required to nullify a threat of an anti-aircraft artillery site. These activities are monitored by the White Force, who have a large array of monitoring equipment. The White Force uses a large display screen in the control room that has computer generated plots of altitudes and flight paths of the Red and Blue Forces, along with specific threat data location and activity. This data is stored and used at the debrief after each mission. The unique feature of the program is that the capability exists to debrief the Blue Forces with the Red Forces, or the simulated enemy and the White Force as a monitor for both air-to-air and air-to-ground threats. To qualify the training benefits to be dervied from the Red Flag Program, an objective point value system has been established. Performance of the aircrew on the last day of the fourteen day training program is compared with that of their performance on the second day of the training program to assess their overall improvement. Results to date have shown an improvement in probability of successful completion of the mission at 37 percent for Air National Guard F-105 squadrons and an overall improvement in the F-4 of 22 percent. The program has been running continuously since late 1975.

MISSION SIMULATOR CONCEPT:

With the ever increasing cost of expandables, such as fuel and special weapons, the Air Force has examined alternates to flying to help maintain mission ready capability. One device that will provide for increased levels of proficiency is the Mission Simulator. Historically, aircrew training devices have been limited to basic instrument flight simulators and emergency procedures trainers. Training in cockpit familiarization was accomplished by using actual aircraft cockpits for switch positioning, etc. With the advent of ever increasing levels of system integration, which has reduced the amount of switchology, we no longer have a viable training transfer with these simple devices. As an example, modern fighter aircraft equipped with a Heads-Up Display (HUD) have essentially placed all flight instruments type data in the direct view of the pilot. It would be extremely difficult to gain proficiency in the operation of the HUD on a basic instrument trainer, and would be extremely expensive and impractical to use an actual aircraft. As aircraft systems become more integrated, the ground based trainers required to support the various aircraft tests are also increasing in complexity. As the complexity increases, the configurations of these equipments also begins to gain a degree of commonality. This complexity and commonality has led to a single piece of equipment that can accommodate one or all of the mission tasks on a sequential basis. The ability to duplicate the mission from engine start to shutdown, with particular emphasis on those mission tasks which are not effectively or safely accomplished in flight, are the objectives of the Mission Simulator.

The current configuration for the Mission Simulator consists of a fixed or limited (60 inch stroke actuators) motion based cockpit platform with a hemispherical visual system wrapped around the cockpit, or cab. The computational system uses a six-degree-of-freedom scheme for the fixed base, with the potential for eliminations of side and longitudinal force for the limited motion base scheme. The current visual systems utilize digitally generated imagery and cathode ray tubes. The simulators use two cabs for interactive aircraft training. This equates to two aircraft mutual support tactics training for air-to-ground. Additionally, the instructor can be a limited participant for two-vs-one air-to-air training. The simulators may use actual aircraft software and related hardware to maintain aircraft and simulator configuration concurrency.

With these configuration features, the degree of fidelity of each mission segment or task that is to be accomplished in the simulator is then defined using several criteria. These criteria are related to safety, cost, geographic constraints, assigned unit capability, individual skills and individual level of proficiency. In the safety related tasks area the simulator can provide more proficiency than actual aircraft training, i.e., out-of-control recovery, multiple emergency procedures and engine shutdown and restart. For all of these tasks, the simulator provies correlation of airframe and environmental cues that the pilot would experience.

In the cost arena, tasks which can, in some cases be more effectively trained for in the simulator are weapons delivery, ECM tactics, and scoring. Such weapons as Maverick and Sparrow missiles are extremely expensive, and as such, are not often launched in a training environment. Additionally, scoring (or lethality) or determining probability of a kill against a simulated threat can be accomplished in the simulator. Threat update data for surface-to-air and air-to-air threats can be incorporated into the simulator more rapidly than actual electronic warfare changes can be modified.

Geographic constraints also adversely impact certain training requirements. Supersonic flight and air space restrictions severely hamper air-to-air training, where speeds range from 200 knots to 1.3 Mach number at altitudes from the surface to 30,000 ft are traversed in one to two minutes. There are also tasks which can only be performed in the simulators, such as training in certain offensive electronic wafare, weapon delivery against moving surface targets, and visual training for defensive threats such as surface-to-air missiles and anti-aircraft artillery. The basic roles of the aircraft and the degree of proficiency required in these tasks define the performance requirements (in terms of fidelity, or

specification material) that are used to define the simulator.

A graphical illustration of how the simulator benefits the training requirements are shown in Figure 4. The two examples show training transfer for terminal aerial gunnery and basic aircraft handling familiarity. The plot shows that, once the basic skills of flying the aircraft are accrued, that a definite payoff exists for terminal gunnery training in the simulator, yet more training transfer can take place in the later task (basic handling) by actually flying the aircraft. The Mission Simulator also provides an additional capability of continuation of training in mission-oriented tasks. Repetitive practices of such tasks in the simulator and then validated in realistic scenarios such as Red Flag provide confidence in readiness capabilities.

Examining the capabilities that the Mission Simulator can provide as a training aid (not a supplement) to a flying program has some rather interesting results. For an air superiority mission wing, we can increase capabilities that would equate to a 21 percent increase in flying requirements by providing a Mission Simulator to assist in training. For an air-to-ground wing, this capability will yield a 22 percent flying equivalent.

In actual dollars for a wing of aircraft, such as the F-16, an annual savings of 7.5 million (air-to-air) and 8.1 million (air-to-ground) dollars can be realized. For an F-16 training wing, an annual savings of 4.6 million can be realized. In terms of time to amortize the investment, this takes place from a high of 7.4 years to a low of 3.6 years. These cost figures are summarized in Table 4.

TABLE 4

SIMULATOR COST SAVING POTENTIAL

	FLYING EQUIVALENTS	DOLLAR SAVED	YEARS TO AMORTIZE INVESTMENT
AIR TO AIR WING	27%	7.5M	3.8
AIR TO GROUND WING	27%	8.1M	3.6
TRAINING WING	9-33%	4.6M	6.8

CONCLUSIONS:

The current objective of the training continuum is to assure that our aircrews are mission ready for the first engagement. The initial and continuing training programs previously described illustrate our trend towards increased emphasis on selection of crew members for the fighter mission, and specialized training for one or more mission tasks. Once a crew member has gained mission ready status on one or more mission tasks, we are placing increased emphasis on specialized training, and crew performance evaluation, such as Red Flag and Mission Simulators. This sepcialized crew training and utilization does allow for increased levels of combat capability (and crew performance) that can be continuously monitored and managed. It also allows for effective management of capabilities in the event of unforseen difficulties, such as excessive maintenance downtime or an aircraft grounding.

With the increased emphasis on mission task capability with existing equipment, two future design considerations can be mentioned. Do we (1) design a multi-mission aircraft with sufficient reliability to be used by several specialized crews in the various mission or (2) design specialized aircraft that are available to train crews to the desired level of proficiency in the various tasks. It appears today that we are tending to use and/or design fighter aircraft with a multi-mission capability. Yet we are training our crews to gain expertise on a task by task basis. The above mentioned design considerations to represent end points of the design spectrum. It appears that, with increasing emphasis on crew specialization, that design trades should include crew training/proficiency in assessing airframe performance and system reliability/maintainability design goals. Such considerations could provide additional design insight from a requirements standpoint to better define the roles of such advanced concepts as supersonic cruise, six degree-of-freedom (control-configured) aircraft, variable cycle engines, etc. Additionally, aircraft availability, which define the reliability and maintainability requirements could also be defined easily in the design process.

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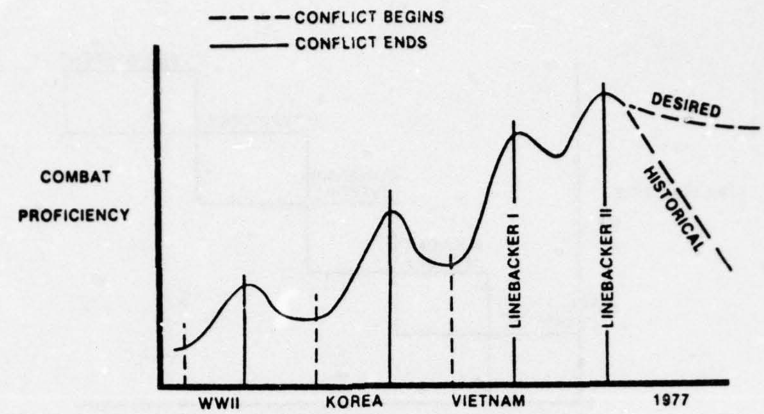

FIGURE 1 COMBAT PROFICIENCY

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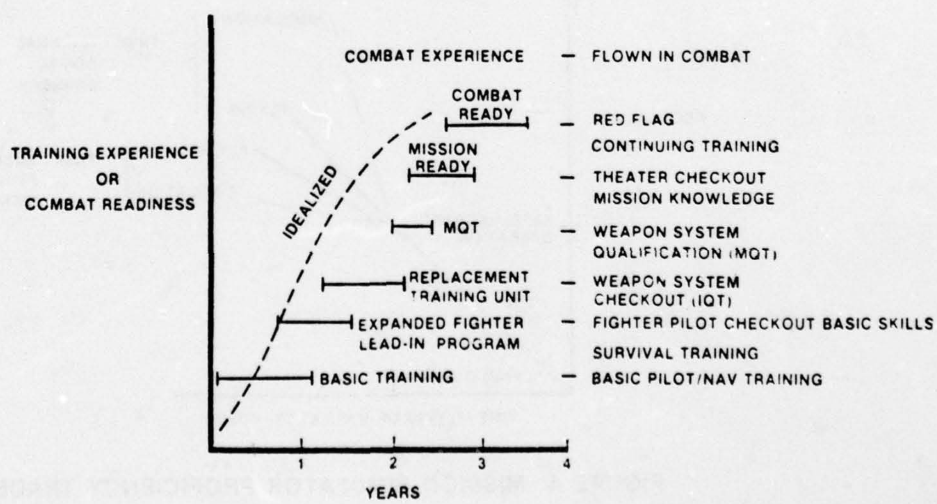

FIGURE 2 THE TRAINING CONTINUUM

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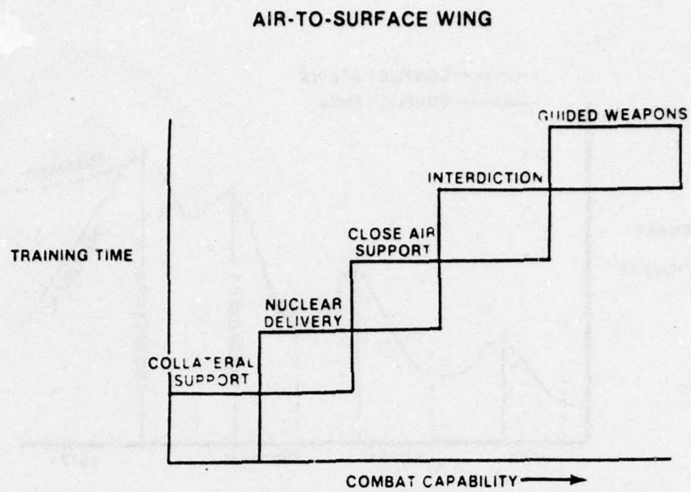

FIGURE 3 GRADUATED COMBAT CAPABILITY

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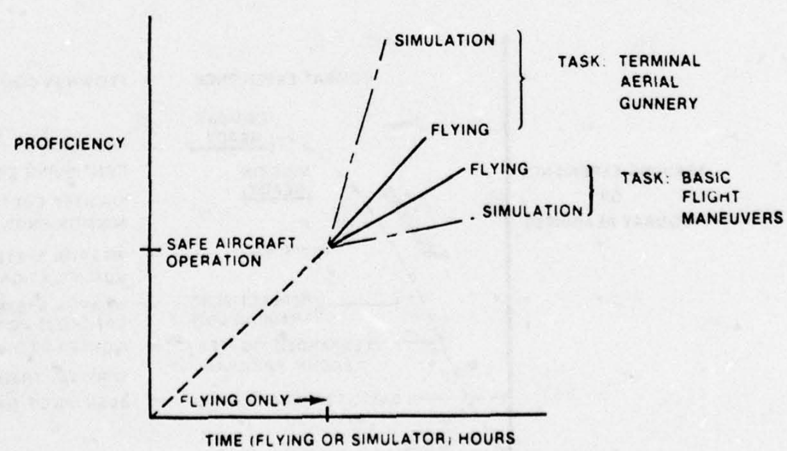

FIGURE 4 MISSION SIMULATOR PROFICIENCY TRADES

LA MISE AU POINT DES ARMEMENTS DANS LA FAMILLE DES MIRAGE

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Les différentes versions de MIRAGE - MIRAGE III, MIRAGE IV, MIRAGE V, MIRAGE F1 et bientôt le MIRAGE 2000 - sont capables d'emporter une grande variété d'armements, qu'il s'agisse de bombes, de roquettes, de missiles air-air ou air-sol, de charges spéciales, de réservoirs de carburant ou de canons internes ou externes.

Cette variété a conduit les A.M.D.-B.A. à développer des méthodes de mise au point et d'essais qui comportent plusieurs phases, depuis le stade initial de l'étude à la planche à dessin jusqu'au stade final des vols opérationnels.

Je vais essayer, bien que n'étant pas spécialiste des armements, mais seulement un ingénieur au Bureau d'Etudes des Avions Prototypes Dassault, de vous faire toucher du doigt quelques aspects caractéristiques de ces étapes successives.

Afin de le rendre moins aride, cet exposé sera suivi d'un court film ; vous voudrez bien excuser la qualité moyenne de ce document qui est un montage de séquences prises en soufflerie, au sol et en vol.

C'est dès le stade de l'étude sur la planche à dessin qu'apparaissent les premières contraintes liées aux armements : lorsque la capacité de l'emport externe représente près de la moitié de la masse au décollage de l'avion lisse, on comprend que les répercussions soient importantes tant du point de vue mise en place, gardes au sol, position du centre de gravité que du point de vue résistance structurale, ou bien encore du point de vue aérodynamique, formes, profils, etc.

C'est ainsi, par exemple, que le MIRAGE F1 a un train d'atterrissage à tube tournant et balancier, de manière que sa cinématique dégage au mieux des volumes importants à la fois sous le fuselage et sous la voilure ; de même, certains réservoirs externes de carburant ont une forme dite "suppositoire", afin que la loi des aires de l'ensemble avion-bidons soit adaptée au mieux.

Puis, viennent les calculs préliminaires de flutter : la détermination des modes vibratoires se fait à partir d'un modèle mathématique dynamique représentant l'avion et ses charges. La principale difficulté rencontrée vient de l'introduction dans le calcul des forces aérodynamiques instationnaires provoquées par les déplacements des différentes surfaces. Ces forces aérodynamiques sont calculées théoriquement à l'aide d'hypothèses linéaires liées à de petits déplacements. Les études actuelles comportent l'introduction des intéractions aérodynamiques entre les surfaces portantes et les charges. Ces hypothèses linéaires ne sont plus valables dans le domaine transsonique avec les écoulements mixtes (en partie subsoniques et en partie supersoniques) et la présence d'ondes de choc qui se déplacent en fonction des déformations. Ce problème qui existe pour une aile pure est aggravé par la présence de charges qui abaissent le nombre de Mach critique.

C'est là que la soufflerie prend le relais : des maquettes dynamiquement semblables sont réalisées et permettent d'étudier le comportement de l'avion dans ce domaine où le calcul s'arrête. Il faut apporter une très grande rigueur à la réalisation de ces maquettes : par exemple, une maquette de MIRAGE à l'échelle du 1/10ème essayée dans la soufflerie de MODANE a une densité équivalente d'environ 0,8. Avec la présence des charges externes, la symétrie des modes doit être particulièrement respectée. Dans la soufflerie, c'est la turbulence de l'écoulement qui crée l'excitation : l'évolution des modes vibratoires et de leurs amortissements est suivie par analyse spectrale. On verra dans le film un exemple.

L'étude des qualités de vol de l'avion équipé de ses charges s'effectue également en soufflerie : stabilité longitudinale et stabilité latérale. C'est par cette méthode que l'on a déterminé préalablement au vol, les limites de centrage arrière du MIRAGE III équipé de 2 RPK, qui sont des réservoirs sous voilure emmenant chacun jusqu'à 4 bombes. L'étude aérodynamique porte également sur la traînée : par exemple, dans le cas du MIRAGE F1, des essais ont été effectués en soufflerie en faisant varier l'écartement frontal des 4 bombes de 400 kg sous le pylône ventral et ont permis de choisir l'entr'axe de traînée minimale.

Au passage, signalons l'existence de combinés d'armement développés par DASSAULT dont on verra des images et qui répondent au même souci de réduction de traînée : tel que le combiné d'emport multiple - roquettes + bombes - ou bien le JL 100 - combiné roquettes + carburant.

Restons dans la soufflerie pour aborder l'étude des séparations : la quasi-totalité des charges externes emportées font l'objet de largages en soufflerie que ce soit à grande vitesse et sous incidence, pour les délestages combat, ou bien à faible vitesse, trains sortis, pour la détresse au décollage.

Les anomalies de comportement au départ sont nombreuses : à cause du champ aérodynamique de l'avion lui-même ou bien des autres charges emportées, on constate des mouvements à piquer ou à cabrer excessifs, ainsi que des mouvements de lacet qui peuvent avoir pour résultat d'une part, de créer un risque de collision, d'autre part, de perturber la balistique et par conséquent de détériorer la précision.

L'arsenal des moyens que DASSAULT a mis au point pour résoudre ces problèmes est très varié : il va de la création de carénages pour obtenir des lois des aires satisfaisantes, jusqu'à des dispositifs mécaniques tels que crochet de retenue arrière, fourchettes d'appui et éjecteur à deux pistons.

Cette technique d'éjecteur à double piston qu' A.M.D.-B.A. a généralisée depuis très longtemps permet de répartir les poussées d'éjection entre l'AV. et l'AR. : ce contrôle de la vitesse de

tangage à l'éjection permet d'éviter les oscillations sur trajectoire.

On verra des exemples dans le film.

Signalons que les éjecteurs retenus présentent une double particularité : afin de réduire les réactions aux attaches, ils sont souvent à double course télescopique ; d'autre part, afin de réduire la traînée, après largage, ils s'effacent automatiquement. DASSAULT développe actuellement des pistons à triple course qui allient une grande vitesse d'éjection à une faible réaction aux attaches.

Tous ces systèmes ont pour but d'obtenir des vitesses verticales homogènes pour toutes les bombes du chargement et d'assurer ainsi les performances opérationnelles de bombardement que permet la grande précision des conduites de tir actuelles.

Les largages mis au point en soufflerie couvrent un domaine de vitesse très étendu : ainsi, l'arme nucléaire du bombardier stratégique MIRAGE IV a pu être larguée jusqu'à Mach 2, au moyen d'une double mise au point, sur les empennages d'une part, sur le système éjecteur d'autre part.

Ne quittons pas la soufflerie avant d'avoir dit un mot des missiles : des pesées sont effectuées sur trajectoire et fournissent les données que l'on introduit dans le modèle de calcul : ce modèle permet de prédire les trajectoires de vol et ainsi d'assurer la sécurité des premiers tirs réels.

Plus tard, lorsque le modèle sera recalé par les essais en vol, on disposera d'un moyen d'étude pour les cas de panne tel que blocage des gouvernes, etc.

Après la première phase d'études sur plan et d'essais en soufflerie, la deuxième étape concerne les essais au sol ; elle revêt deux aspects :

- l'avion terminé et équipé de ses charges externes est soumis à des essais de vibrations qui ont pour but de déterminer, en vraie grandeur, les modes vibratoires. Pour ces essais, l'avion est pendu dans des élastiques. Les mesures faites permettent de recaler les calculs de flutter.
- parallèlement, les systèmes d'éjection font l'objet de la mise au point finale avant vol : ces réglages s'effectuent sous portique, d'abord, sous avion ensuite. On ajuste les vitesses angulaires de tangage. On vérifie les réactions aux attaches, de même que la réponse de la structure sous l'impulsion de départ.

Enfin vient la dernière phase de la mise au point, c'est-à-dire les essais en vol.

Il s'agit d'abord d'ouvrir les domaines dans les différentes configurations retenues : DASSAULT a développé une méthode d'exploitation en temps réel des modes vibratoires et de leurs amortissements. L'avion est équipé d'un excitateur : exemple, dans le cas du MIRAGE F1, la servo-commande d'aileron est pilotée par un signal soit bruit blanc, soit balayage sinusoïdal : on connait ainsi le signal d'entrée : les sorties sont mesurées par des capteurs accélérométriques.

Ce dispositif présente l'intérêt d'être utilisable quelles que soient les charges externes emportées.

Au cours des vols d'emport, on procède également aux mesures d'efforts ; à l'aide des maquettes représentatives des formes externes équipées de balances de pesée, on enregistre les efforts introduits lors des différentes manoeuvres.

- Ensuite,ont lieu les largages et les tirs.

Malgré toute l'aide apportée par les essais préliminaires en soufflerie, les essais en vraie grandeur font apparaître la nécessité de mises en point complémentaires : ainsi, dans le cas des largages de bombes sous le MIRAGE F1, il a fallu choisir de larguer sous voilure d'abord les bombes internes avant les bombes externes pour obtenir des trajectoires de départ satisfaisantes. Il a été également nécessaire de rajouter des appuis arrière afin de corriger le mouvement initial de remontée des bombes, lors de bombardement en piqué pur.

Quelques exemples seront montrés dans le film.

Pour les tirs de missiles, un des problèmes à résoudre concerne l'effet du sillage chaud des propulseurs sur le comportement des entrées d'air et de la manche. Afin de ne pas affecter le fonctionnement du moteur, un système appauvrisseur de tir a été mis au point : il a pour but, pendant le court instant du tir, de diminuer automatiquement la charge du réacteur par réduction du débit carburant délivré. l'appauvrissement est réglable ; il est par exemple différent dans le cas du tir au canon et dans le cas du tir des missiles soit au coup par coup, soit en salve.

Ainsi, au travers de ces quelques exemples, depuis les études préliminaires jusqu'aux essais en vol en passant par les essais en soufflerie et au sol, vous avez pu vous faire une première idée de la variété, mais aussi de la complexité des problèmes posés par l'intégration des armements aux avions.

Dans le cas de la famille des MIRAGE, la mise au point à laquelle DASSAULT travaille depuis bien des années a fait apparaître des solutions originales.

Pour compléter ce résumé rapide, voici quelques compléments illustrés dans le film que vous allez voir maintenant.

AIR COMBAT

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SUMMARY

The analysis technique described by this paper provides a solution to the generalized multiple component engagement of opposing military forces and employs a complimentary blend of theory and practice. The results have the advantage of providing detailed insights through the use of asymptotic models, human interactive analysis, and topological arguments.

The paper demonstrates the technique by example, using the fighter escort problem. Application to any air combat problem of reasonable dimension is possible. Aircraft performance, avionics, and weaponry are considered. The results can be used to guide the allocation of R&D resources, to evaluate the force-mix problem related to new weapon system purchases, and to advise operational commands of deployment and tactics for known forces.

INTRODUCTION

The central problem in the development of a nation's power is what mix of weapon system types to choose for the force structure, including what the specifications on aircraft performance, weapons complement, and avionics suite for each component of the mix should be. In addition to the identification of weapon system types for the mix, military and government planners and decision makers must also deal with the question of numbers; that is, how many of each weapon system type to acquire.

The purpose of this paper is to sketch a part of the methodology of a powerful technique for treating this important problem and to present some results which are of interest.

The force structure problem is a broad one. It is clearly a problem imbedded in the larger and still more complex problem of constructing a national defense policy which must necessarily be concerned with national goals and how they are to be attained through political and economic means. These considerations are not addressed herein.

To bound this discussion and make it possible to clearly illustrate the power of this technique, the focus will be on the simpler task of determining the desirable weapon system characteristics of an escort fighter whose mission is to defend a formation of strike aircraft against attacks by enemy interceptors. Also to be determined is the number of escorts required for an adequate defense.

THE ESCORT PROBLEM

For clarity then, a restricted problem in air combat is treated; the escort problem. Many considerations, including scenario-dependent ones, such as AAA and SAM defensive networks and availability to either side of airborne warning and command and control systems are omitted, but can be folded back in. Figures 1A and 1B illustrate the basic character of the escort problem; namely, one or more escort fighters attempt to establish an impregnable defense of the strike formation. This can be accomplished by escort fighters stationing themselves in relation to the strike formation so that an impermeable detection surface is formed. An impermeable detection surface is created when enemy interceptors, attacking singly from any direction or in masses from several directions simultaneously, cannot reach a firing zone where they can successfully and safely launch ordnance at members of the strike formation without first being detected at the surface in time for the escorts to identify them visually and counterattack.

THE STRIKE FORMATION

A strike formation is classically often a carefully structured array of several aircraft. The configuration and formation tactics selected must provide, in general, for survival against not only interceptor attacks (if escorts are not employed), but also AAA barrages and SAM salvos. As a result, the strike formation usually must maintain its integrity, and aircraft comprising it are severely limited in their freedom to move about autonomously. Other mission demands, such as fuel conservation and the weight and drag burden of payload including external stores, electronic countermeasure pods, etc. further

limit the maneuverability of strike aircraft. Because of contraints like these, it is seldom that strike aircraft can make a major contribution to their defense through formation maneuvers. Since they are normally mission-defeating, defensive measures such as jettisoning payload and/or breaking-up the formation are not considered practical means of defending a strike. Therefore, for the purpose of this paper, it is reasonable to assume that the strike formation can be treated as a unit moving at nearly constant speed, heading, and altitude. Thus, defense of the strike against enemy interceptors is the sole responsibility of the escort fighters.

In the figures to follow, the strike formation is represented by the silhouette of the single aircraft. The strike formation is approximately one nautical mile in depth and 0.75 nautical miles in width. The aircraft silhouette used to symbolize the strike formation is always scaled to these maximum dimensions. Values selected for the strike formation's cruise speed and altitude are typical of today's fighter-bomber designs, specifically, 0.8 Mach and 17,500 feet, respectively.

Letting the values of cruise speed and altitude of the strike formation approach zero is equivalent to transforming the escort problem into the combat air patrol problem or in the limit to the air defense problem. Removing defense of the strike from consideration entirely converts the escort role from defense to offense and is equivalent to transforming the escort problem into the fighter sweep problem. By adapting the technique described in this paper, any of these transformations can be handled.

THE ENEMY INTERCEPTOR

A hypothetical fighter (see Table 1) is used for an example of an enemy interceptor. The airframe (aerodynamic and structural) and propulsion characteristics of this aircraft are readily summarized through the use of conventional energy-maneuverability and load factor versus speed (V-n) diagrams, see figures 2 and 3.

TABLE 1

INTERCEPTOR CHARACTERISTICS AT 17,500 FEET ALTITUDE

* Placard Speed	1.5 Mach
* Turn	5.0 G at 1.5 Mach
* All-Aspect Missile	7.0 nm range at 1.5 Mach launch
* Rear-Aspect Missile	7.0 nm range at 1.5 Mach launch Boresight launch only Aspect angle within ± 45 degrees
* Gun	Zero effectiveness outside 2000 feet range or ± 30 degrees angle-off

Note - Missile range is ground distance from launch to point where missile slows to zero speed relative to interceptor.

The characteristics of the interceptor's weaponry and fire control system can be summarized suitably in the form of firing zones or launch envelopes. For simplification, it is assumed that the interceptor is under ground control so airborne detection means for vectoring to and positioning on the target are not employed. With these characterizations of the enemy interceptor available, it is possible to describe its capability against the strike formation.

The description of enemy interceptor capability appearing to be most satisfactory for analyzing this restricted problem in air combat is shown in figure 4. For three different types of air-to-air weaponry in use today, the loci of equal times-to-fire are displayed. The coordinate frame used is fixed to and moves with the strike formation. These loci are constructed on the assumption that the preferred strategy for the enemy interceptor is to attack along a minimum time trajectory. While it has not been possible to rigorously establish yet whether or not this strategy represents optimal play in a differential game sense, heuristic arguments suggest it is. Intuitively, the difficulty of defending against an interceptor increases as the swiftness of the attack increases. Historical evidence also supports this assumption. For example, the ME 262 in World War II, the MIG-15 in Korea and the MIG-21 in South Vietnam were all able to successfully exploit their maximum speed intercepting B-17s, B-29s, and F-105s respectively, which were escorted in turn by F-51s, F-86s, and F-4s. Unless the escorts can establish an impermeable detection surface, it is also preferable for the interceptors to ignore the escorts and concentrate their attacks on the strike formation. Again, there are heuristic arguments and historical evidence to adequately support this choice of strategy.

Some obvious, but very relevant features of these loci should be noted and discussed. First, if the interceptor elects or is constrained to fire rear-aspect weapons, its avenues

of approach for a successful engagement of the strike formation are limited. It can be seen from figure 4 that attacks approach along trajectories which converge toward a relatively small region, on the order of a nautical mile or two, aft of the strike formation. In contrast, interceptors launching all-aspect weapons enjoy a significantly greater freedom in their choice of minimum-time trajectories along which to attack. The figure shows the broadness of the region in which successful launch opportunities exist. Second and very important, the ratio of the maximum speed of the interceptor to the cruise speed of the strike formation is the parameter largely determining the spacing between contours of equal time-to-fire. As the value of this parameter is increased, the spacing widens. In general, this widening translates into a reduction in the time available for escorts to react and develop their counterattack. Third, interceptors conducting head-on attacks from above have the advantage of being furthest from the strike formation for a given time-to-fire. Why this is an advantage becomes apparent when escort detection of the incoming attack is taken into account.

In summary, it should be recognized that the figures presented in this section graphically describe the capabilities of the enemy interceptor in combination with its weaponry. All of their physical properties and operational limitations (human factors excepted) are thereby given a geometrical interpretation. This is extremely useful in understanding and explaining the four-dimensional, space and time, character of air combat. That is the reason for their employment and for making some topographical observations about the nature of these loci.

THE ESCORT

As was done in the case of the enemy interceptor, a hypothetical fighter is used for an example of an escort having capabilities representative of today's models (see Table 2).

TABLE 2

ESCORT CHARACTERISTICS AT 17,500 FEET ALTITUDE

* Cruise Speed	0.9 Mach
* Placard Speed	1.8 Mach
* Turn	7.0 G at 0.9 Mach
* Acceleration	0.75 G at 1.0 G
* Combat Fuel	5.0 minutes
* All-aspect Missile	20 nm range at 1.8 Mach launch 15 nm range at 0.9 Mach launch ± 45 degrees off-boresight
* Rear-aspect Missile	17 nm range at 1.8 Mach launch 15 nm range at 0.9 Mach launch ± 45 degrees off-boresight Aspect angle within ± 45 degrees
* Gun	Zero effectiveness outside 2000 feet range or ± 30 degrees angle-off

Note - Missile range is ground distance from launch to point where missile slows to zero speed relative to strike.

Again, the characterization of this hypothetical fighter and its weaponry is facilitated thru the use of energy-maneuverability and V-n diagrams and a launch envelope. A comparison of the escort with the interceptor, if accomplished in the traditional manner of overlaying their energy-maneuverability contours, would suggest that the escort is the more "capable" aircraft. It will be seen in the next section why such a technique for evaluation of weapon systems can be so misleading.

For the restricted problem in air combat selected for analysis in this paper, the role of the escorts is to prevent the interceptors from reaching a position to fire upon the strike formation. This is the "right" strategy for the escort to pursue under the assumption that the preferred strategy for the interceptor is to attack the strike formation and not the escort. Again, the constraints of aircraft aerodynamic, structural and propulsive capabilities and weapon envelope limit the escort's choice of counterattack options. In addition, the escorts must consider the number of interceptors attacking. If they are outnumbered, then an impermeable barrier can be constructed only if some subset of the escorts can re-attack one or more times.

To appreciate the escort's capability against an interceptor attack a geometrical description of its maximum positional mobility relative to the strike formation is employed. The escort's "footprint", shown to different scales in figures 5 and 6, is by definition the loci of most distant positions reachable in equal times. Escorts executing the

minimum time maneuvers required to establish this footprint will normally fly in afterburner. Combat in afterburner is generally limited to just a few minutes for contemporary fighters so this is a severe limitation on the escort's exploitation of its maximum positional mobility.

Some interesting features of the escorts footprint deserve mention, since they will tend to dominate the solution of the escort problem. First, in a short time, on the order of a minute or less, an escort can move forward of the strike formation only a few nautical miles and rearward a comparable amount, see figure 5. Second, as the time increases beyond one minute or so, an escort can move rearward of the strike formation much further than forward. Third, to move forward of the strike formation a distance on the order of fifty nautical miles, an escort would consume its combat fuel allowance, see figure 6.

It becomes apparent from studying these figures, first, that the further from the strike formation it is necessary for an escort to engage an interceptor, the more time and fuel is required to attain a firing position. Second, if escorts must engage more than once, when they are outnumbered for example, then there may be either insufficient fuel or time for conducting the second or subsequent counterattacks.

To some extent the capability of the avionics suite of the escort has already been described implicitly by the launch envelopes of its weaponry, figure 7. The feature remaining to be considered is the means of target acquisition by the escort. It will be assumed that the escort can detect interceptors with complete certainty at the instant they cross the boundary of the escort's detection envelope, they model of which is a cone with a sixty degree half angle. The maximum detection range is not specified since it and the number of escorts will be treated as the independent and dependent variables respectively in the analysis which follows.

Ultimately, it will also be required of the escort to make all identifications visually. It will be assumed for simplicity that identification coverage is isotropic and positive identification is possible whenever an interceptor is within three nautical miles of an escort.

DEFENSE OF THE STRIKE FORMATION

Thus far, strike formation, interceptor attack and escort counterattack capabilities have been described asymptotically entirely by geometrical interpretations of aircraft, weapon and avionics performance. These geometrical interpretations are extremely useful in developing the topology of air combat, providing at the same time, not only a tool for analysis, but also a simple mechanism for explanation. The figures presented in the previous sections may be thought of as the air combat analogy of describing the permissible moves of chess pieces.

It remains to discover interceptor attack and escort counterattack tactics which represent for each side the "best" offense and defense. These tactics can be constructed by means of an algorithm, which is explained with the help of a series of figures (8A thru 8C). These figures represent the significant events during an encounter between a single escort and a single interceptor. Note the change in scale between figure 8A and the remaining figures in the series. Note that this analysis only considers two dimensional engagements confined to the horizontal plane of the strike formation. Extensions to three dimensions follows a similar but lengthier procedure. Extension to include more than one fighter on each side is treated in a subsequent section.

In figure 8A an escort is seen initially at its assumed position relative to the strike formation. At this same moment an interceptor attacking from an arbitrary direction arrives at the detection surface causing the escort to initiate a counterattack. As seen from the loci of equal times-to-fire, the interceptor can launch an all-aspect missile at the strike formation about 100 seconds after the engagement starts.

Figure 8B shows the position of the aircraft at the time the escort can first fire its all-aspect missile against the interceptor. This event, for the initial conditions selected, occurs before the interceptor has its opportunity to open fire. As seen in the figure, this occurrence is determined from an overlay of the escort's footprint and launch envelope.

The final figure in this series, figure 8C, shows that the escort missile hits the interceptor before the interceptor can launch its own weapon. This condition is used in this paper to define an impregnable defense. No time delays for pilots or equipment have been factored into this specific example, but such considerations can be included with minor modification to the procedure.

The algorithm employed in this analysis has two iterative loops. The inner loop searches for the set of initial positions at which the escort can be stationed so that it can conduct a successful counterattack regardless of the direction of the interceptor's attack. The outer loop searches this set of stations to find the subset for which the maximum detection range is a minimum. When these stations exist and can be found, it is believed that escort and interceptor are employing optimal tactics for the "game" as it has been defined.

If the set of stations is empty, the escorts capability is inadequate for defense of the strike against a single interceptor. To remedy this situation, either the escorts overall capability must be improved, or more than one escort must be employed, or some combination of both. For the escort and interceptor characteristics selected in this example it is found that a single escort weaving laterally slightly about a station just aft of the strike formation constitutes an impregnable defense against a single interceptor attack, provided the maximum detection range is not less that approximately fifty nautical miles, see figures 9 and 10.

If single interceptor attacks are spaced no less than seventy-five seconds apart, then the escort will have sufficient time to complete one engagement and reposition itself at the desired station before the follow-on engagement begins. Fuel and armament loads obviously limit the number of engagements a single interceptor can conduct. In this example, if interceptor attacks are repeated from a head-on position, the escort would run out of fuel after no more than three engagements. If the probability of kill in a head-on attack is not high enough and additional time must be provided to allow the escort to turn, pursue, and fire upon the escort from the stern, then the maximum detection range would have to be increased about ten nautical miles. A five nautical mile increase would provide the escorts approximately ten more seconds to set up the counterattack.

Also in this example of a single escort counterattacking a single interceptor, the best choice of position for the escort station is determined by interceptor attacks from the stern. Thes escort must station itself far enough aft of the strike formation to allow a successful counterattack based on visual detection. The greater the interceptors overtake speed in a stern pursuit of the strike formation, the further aft the escort must be stationed. Equipping the escort with a means of detecting aircraft approaching from its rear, rather than depending on detecting visually, greatly alleviates this situation. The maximum detection range is then driven by the requirement for timely detection of head-on attacks by the escort stationed in this position. First escort deployment characteristics are summarized in Table 3.

TABLE 3

FIRST ESCORT DEPLOYMENT CHARACTERISTICS

* Extensive Use of Positional Mobility
* Rear- and All-Aspect Missiles Employed
* Minimum Number of Escorts - One
* BVR Requirement About 50 NM for Single Interceptor Attacks
    - Add 5 NM for Every 10 Seconds Added Warning Time
    - Add 10 NM for Re-Attack Time
* Escorts Vulnerable to Attacks from Rear
* Re-Engagement Difficult if Time Between Attacks Short
* BVR Requirement About 100 NM for Massed Interceptor Attacks if Escort Numbers Equal Interceptor Numbers

From the analysis just made, the weakness of the escort's defense should be apparent. Aside from the difficulties of detecting and identifying interceptors, the time and fuel required to move forward of the strike formation compounds the difficulty of the defense. This problem stems from the fact that the escort cruises at the same subsonic speed as the strike formation. Even if the escort could accelerate to its supersonic dash capability almost instantly, the difficulty would not be removed entirely.

Obviously the greatest challenge is in the defense against more than one interceptor, a problem not easily solved. When interceptors mass for a simultaneous attack, an increase in escorts positioned at the aft station does not solve the problem alone, even if the number of escorts equals the number of interceptors. The reason arises from the fact that the number of interceptors in a massed attack cannot be resolved at the time of detection because of the physical limitations of contemporary sensors. In addition, visual identification must be observed as a rule of engagement.

Since the number of interceptors is not immediately known, the number of escorts to commit to a counterattack is uncertain. Allocating the incorrect number of escorts has serious consequences. If too many are committed, then the number left on station (if any) is drawn down unnecessarily, weakening the potential defense against attacks coming from other sectors. If too few are committed, then some of the interceptors may avoid engagement by these and hit the strike formation before those escorts in reserve can counterattack successfully.

A solution to this problem, illustrated in figure 11 involves the dispatch of a flight (two to four fighters) of escorts in advance of the main counterattack. This flight is *responsible for identifying the interceptors, determining their number*, and calling for support. Although this tactic works, the timing of the main counterattack requires that the maximum detection range be increased to one hundred nautical miles. Furthermore, nearly all of the escorts combat fuel allowance is exhausted executing this maneuver.

To search for ways to overcome the shortcomings of the two defense tactics just described, the algorithm is exercised again for smaller values of the maximum detection range, on the order of the distance being used for the visual detection range. For these values, a single escort does not have enough warning time to counter single interceptor attacks inbound from certain directions. To construct an impermeable detection barrier a number of escorts must be developed in a pattern such as illustrated in figure 12. For this case detection provides an escort just enough time to turn toward the incoming interceptor and fire. Minimal use is made of the escorts positional mobility during the counterattack, since the escorts missile footprint provides the necessary reach.

Although stationed in this configuration the escorts can counter interceptors attacking singly, the second deployment is still not satisfactory with respect to countering massed attacks. However, the second deployment is better than the first with respect to re-engagement potential. Counterattacks are much shorter in duration, so less fuel is consumed and displacements from the original stations are not appreciable. Therefore, deployed this way, the escorts can accomodate significantly shorter intervals between a series of single interceptor attacks. However, under operational conditions it would be impractical to deploy escorts in this manner because setting-up and maintaining this configuration of stations is too involved. Second escort deployment characteristics are summarized in Table 4.

TABLE 4

SECOND ESCORT DEPLOYMENT CHARACTERISTICS

* Minimal Use of Positional Mobility
* Rear- and All-Aspect Missiles Employed
* Ten Escorts to Man All Stations
    - Station Keeping Difficult
* Almost No BVR Requirement for Single Interceptor Attacks
* Some Escorts Vulnerable to Attacks From Rear
* Re-Engagement Greatly Facilitated
* Escorts Cannot Counter Massed Interceptor Attacks

Two deployments of fighters have been constructed (refer to figures 10 and 12), which illustrate some interesting features of the escort problem including a trade-off between maximum detection range and number of stations. In the situation of a defense against interceptors attacking singly, these deployments represent the limiting cases of this trade, since the first deployment requires the least number of stations and the second deployment requires the least maximum detection range. Between the limits defined by these two cases, a family of discrete deployments can be constructed. Any deployment in this family will require shorter maximum detection range than the first case, and fewer stations than the second case.

For example, an intermediate case has been constructed for which the maximum detection range is twenty-five nautical miles and the number of escort stations is two, see figure 13. This deployment allows for an appreciable reduction in the maximum detection range by making use of the forward-stationed escort to handle the head-on attacks. Station keeping is easier. Therefore, it appears to be an "attractive" compromise between the two deployments which are the limiting cases.

Having examined the special situation of interceptors attacking singly the more general case of interceptors attacking in mass is turned to next. A modification of a defense against single interceptor attacks, the tactic illustrated in figure 11 provides valuable insight to the treatment of massed attacks. From this tactic it is learned that resolving the number of interceptors in a massed attack is the first problem to solve in the construction of an impregnable defense. Once this is accomplished the efficient allocation of escorts for a counterattack can proceed. It is also learned that unless additional time is provided for re-engagements (usually at the expense of increased maximum detection range) the number of escorts must equal the number of interceptors in any given attack. From the intermediate case deployment, the tactic illustrated in figure 13, it is learned that stations forward of the strike formation are efficient, with respect to required maximum detection range and fuel or time used, in dealing with head-on attacks. Based on these observations a defense is constructed which minimizes the

number of escort stations and the maximum detection range jointly, given that no more than one engagement per escort is permitted. Any other defense requires either more stations or longer maximum detection range or both. This deployment is illustrated in figure 14 and the defensive tactic associated with it is shown in figure 15. In the construction of this defense, the assumption is that the number of interceptors in a massed attack is not resolved by the sensor until they approach to within ten nautical miles of it.

In all of the deployments described, aft-most stationed escorts (because reliance is on visual detection) are in a marginal situation with respect to self-defense against interceptor attacks approaching from the rear. To eliminate this vulnerability, by improving detection coverage rearward solely thru the employment of the given escort sensor capability, would require the escorts to orbit in some complicated manner (see figure 16) because of the geometry of the detection envelope. A possible alternative to orbiting, which has some drawbacks, is equipping escorts with sensors having a rearward view. Another alternative is the employment of an airborne warning system like AWACS. By whatever means, rearward coverage beyond visual detection range is a must for escort self-defense. If detection occurs at about twenty nautical miles, then a massed interceptor attack from the rear is countered from the fourth deployment as shown in figure 17. Fourth escort deployment characteristics are summarized in Table 5.

TABLE 5

FOURTH ESCORT DEPLOYMENT CHARACTERISTICS

* Countering Single Interceptor Attacks
  - Limited Use of Positional Mobility
  - All-Aspect Weapons Not Required
  - Time Available for Re-Attacks
  - Re-Engagements Facilitated
  - BVR Requirement About 25 NM
* Countering Massed Interceptor Attacks
  - Positional Mobility Used Extensively
  - All-Aspect Weapons Not Required
  - BVR Requirement About 25 NM
* In General
  - Aft Stationed Escorts Vulnerable to Attacks From Rear
  - Station Keeping Not Difficult

CONCLUSION

A restricted problem in air combat; a simplified escort problem, has been analyzed by means of a technique which builds on a blend of theory and practice.

The technique and some results of its application have been described largely in geometrical terms which is natural for developing an understanding of the topology of problems in air combat.

The results presented help illuminate the nature of the strong interaction between a weapon system type and the number of that type required. An appreciation for the effects of changes in the weapon system specifications has been obtained also.

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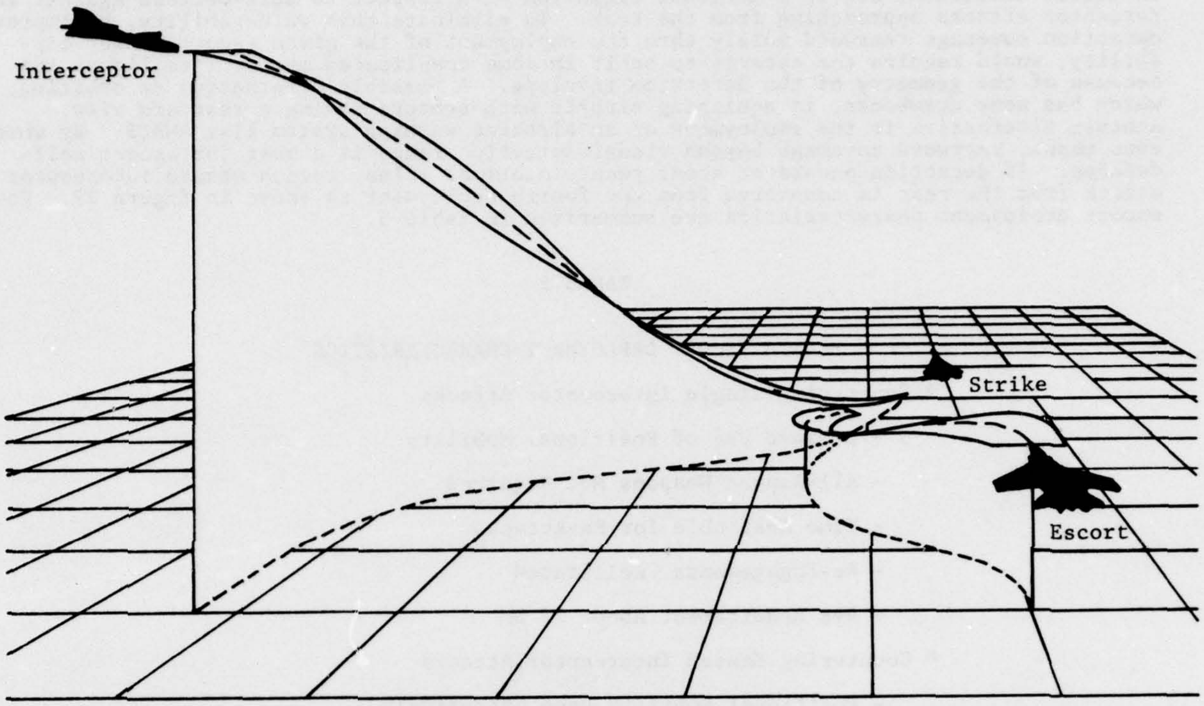

Figure 1A; The Escort Problem: A Typical Rear Hemisphere Engagement

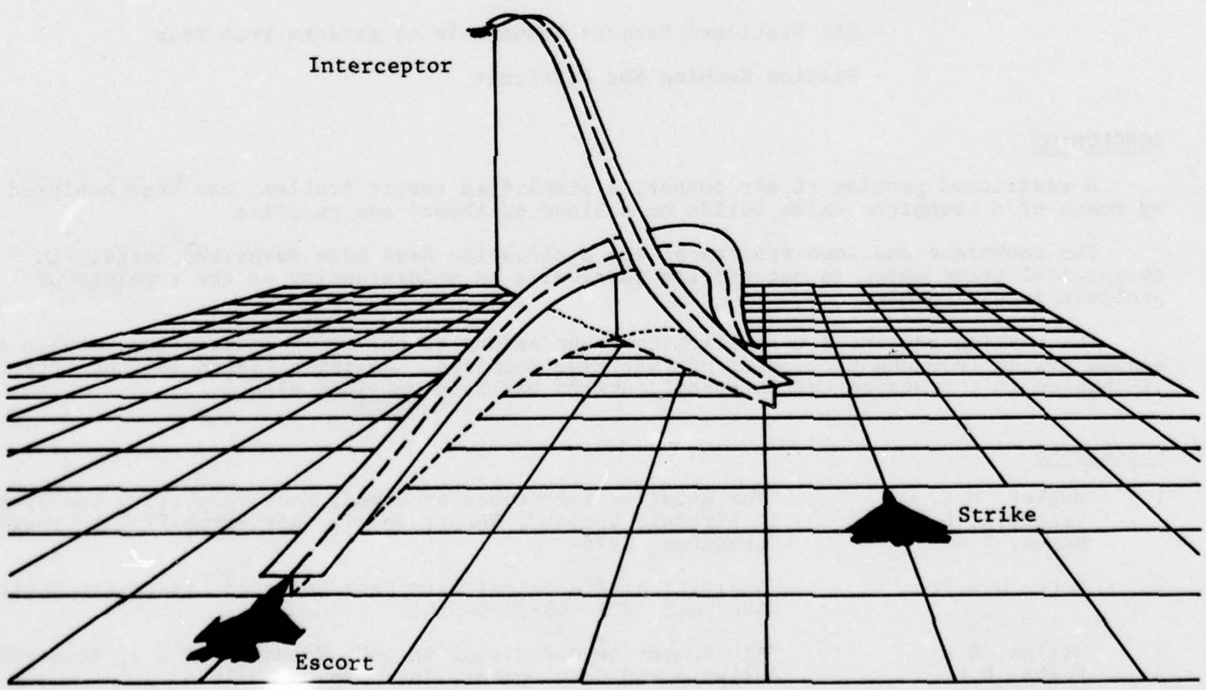

Figure 1B; The Escort Problem: A Typical Front Hemisphere Engagement

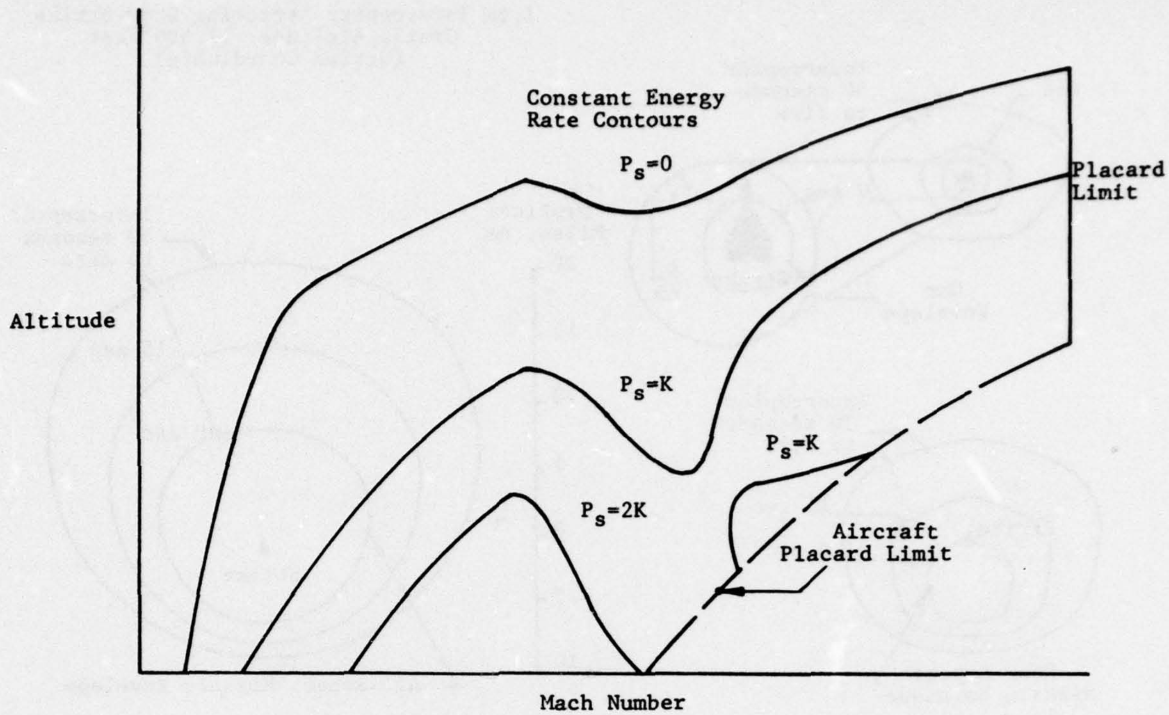

Figure 2; Typical Energy Maneuverability Plot

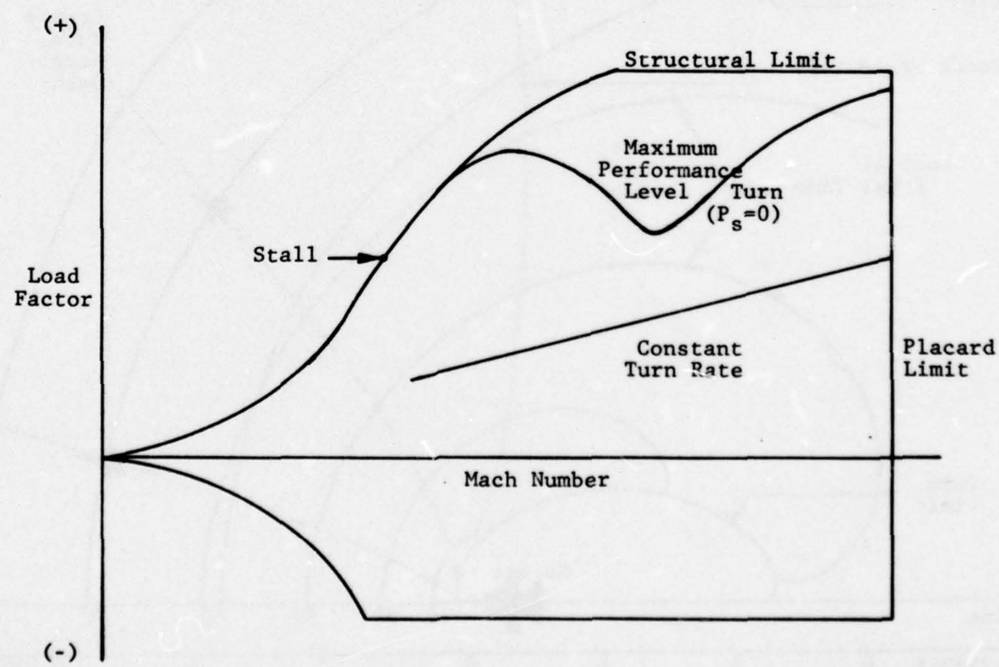

Figure 3; Typical V-n Diagram

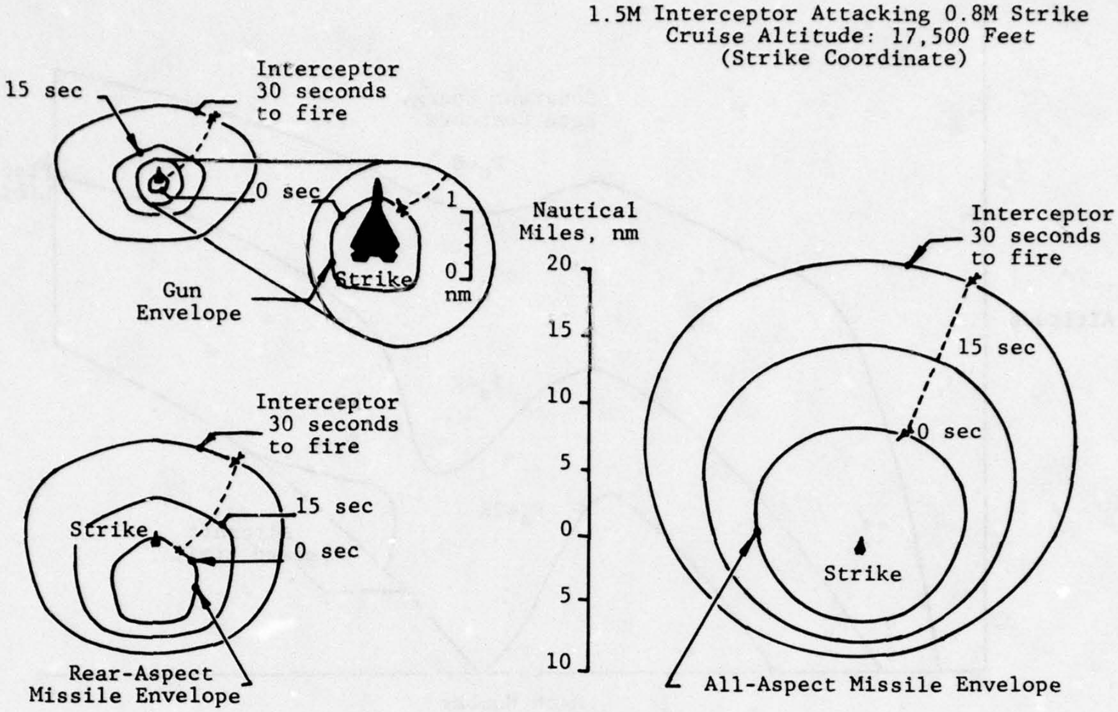

Figure 4; Loci of Interceptor Equal Times-to-Fire

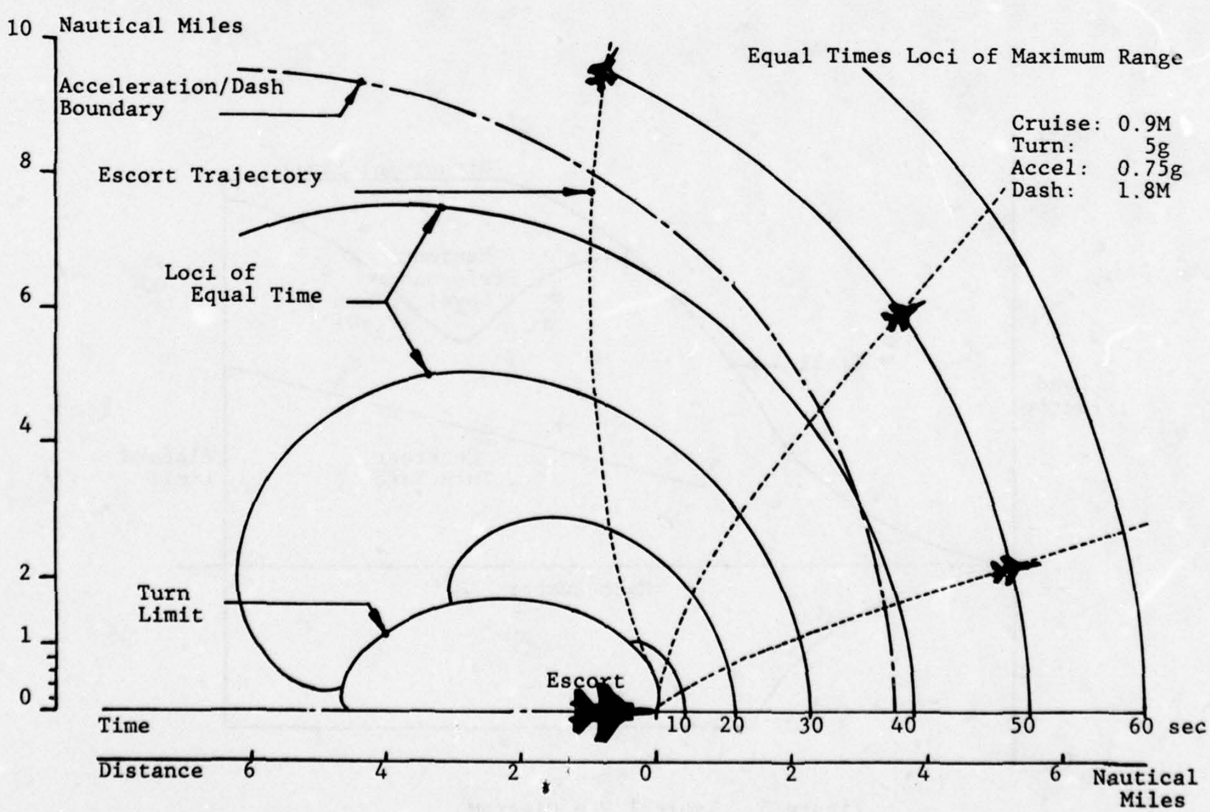

Figure 5; Escort Footprint

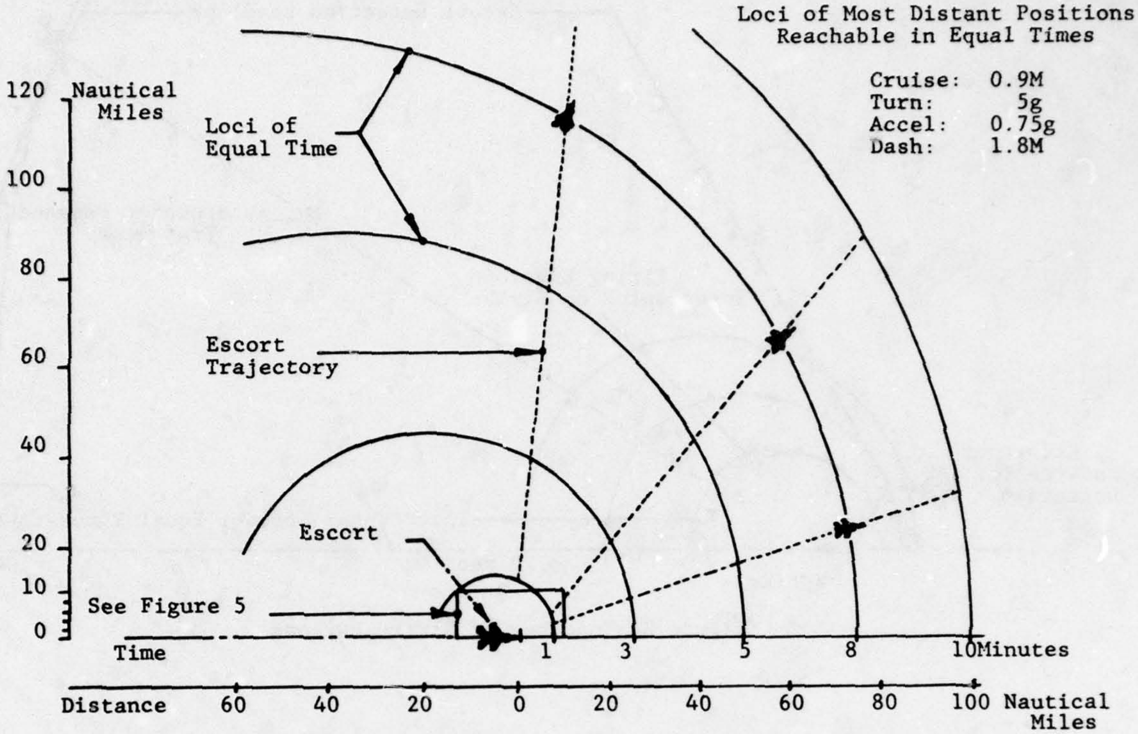

Figure 6; Escort Footprint

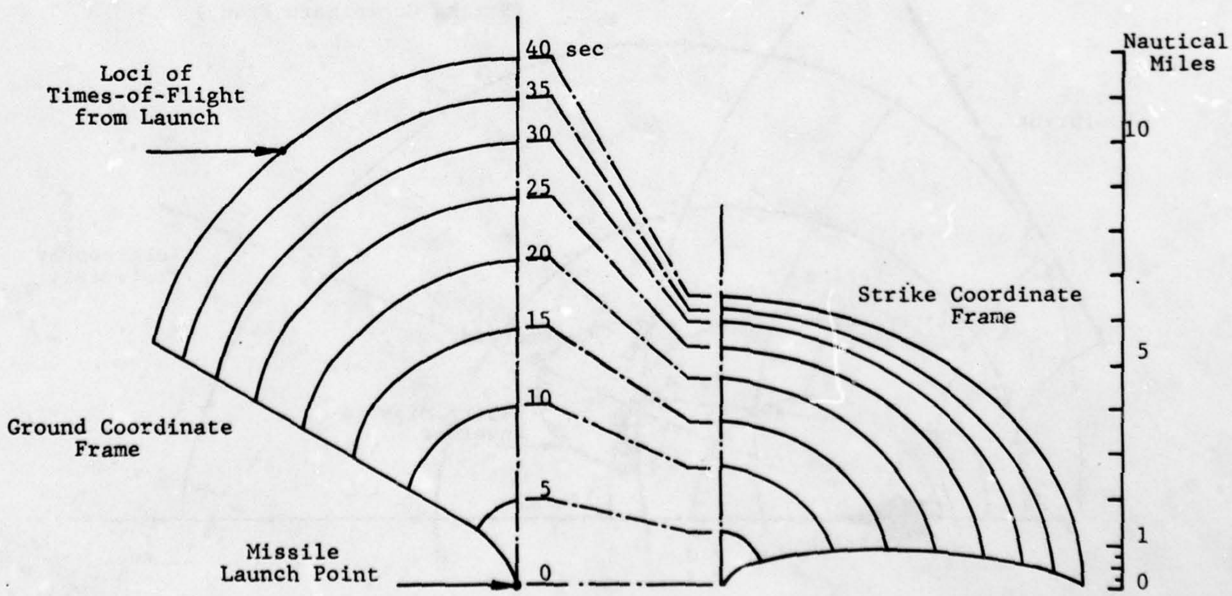

Figure 7; Escort All-Aspect Missile Footprint

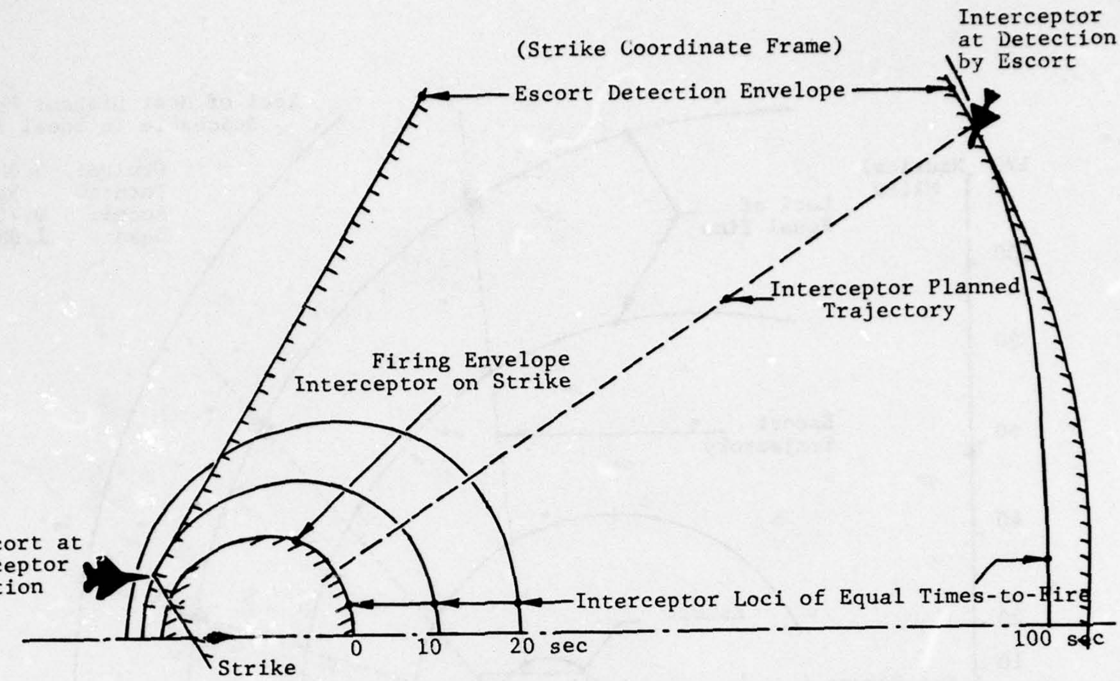

Figure 8A Topology of Strike Defense

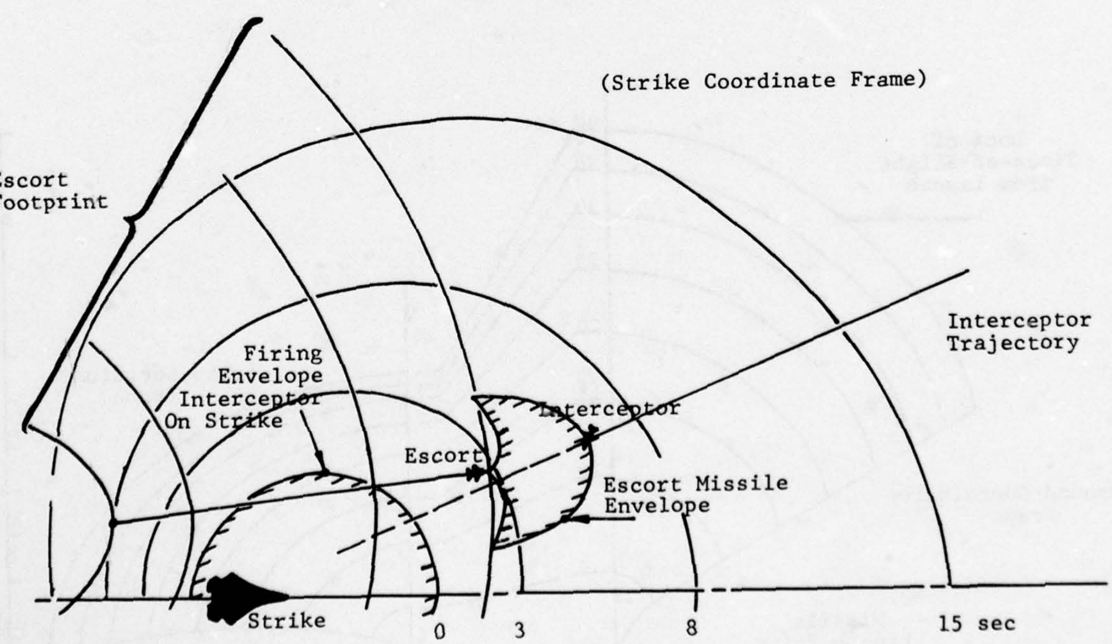

Figure 8B; Topology of Strike Defense

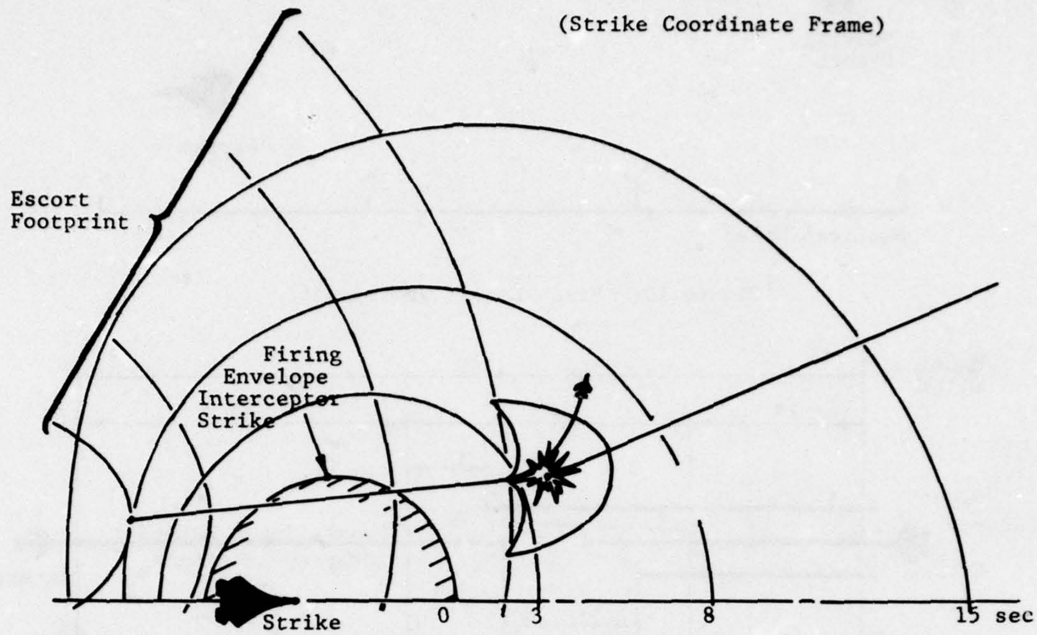

Figure 8C; Topology of Strike Defense

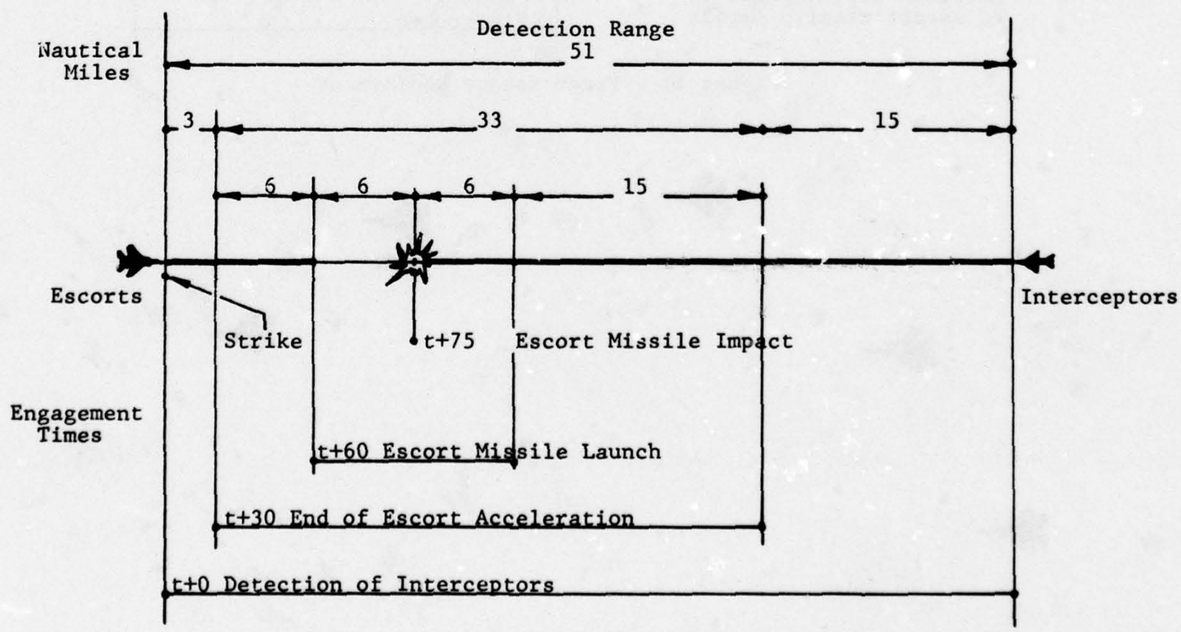

Figure 9; First Escort Deployment Construction

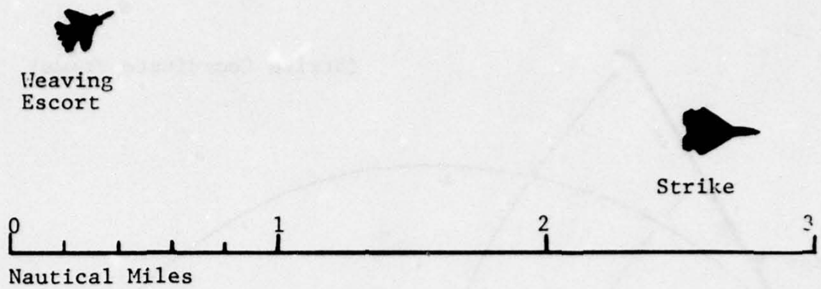

Figure 10; First Escort Deployment

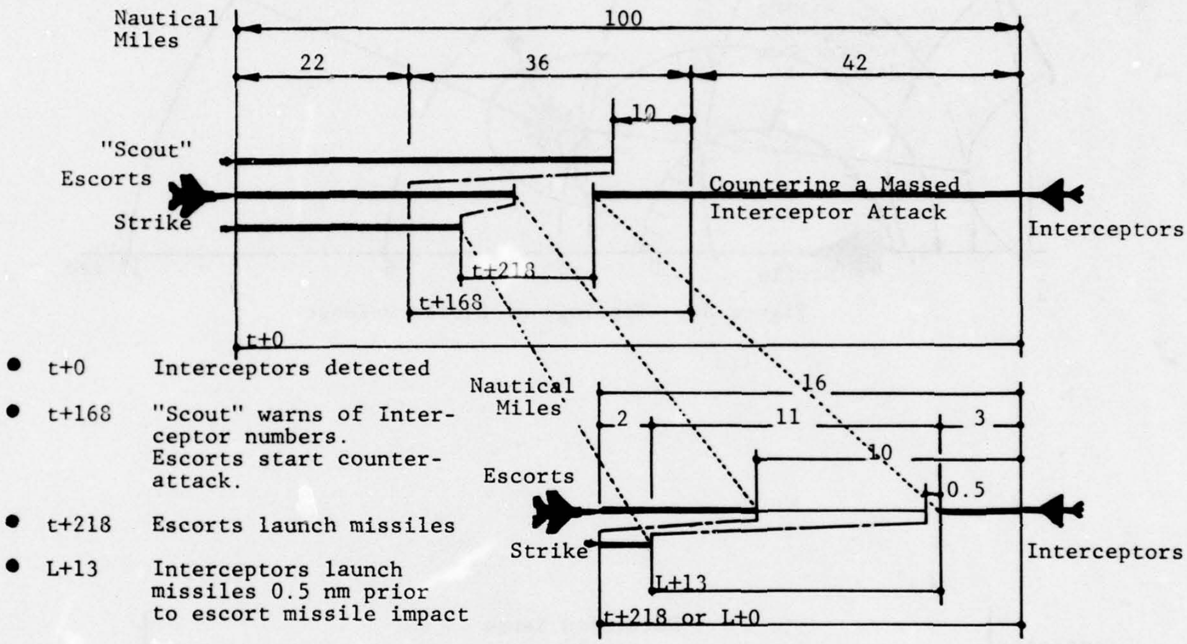

Figure 11; First Escort Deployment

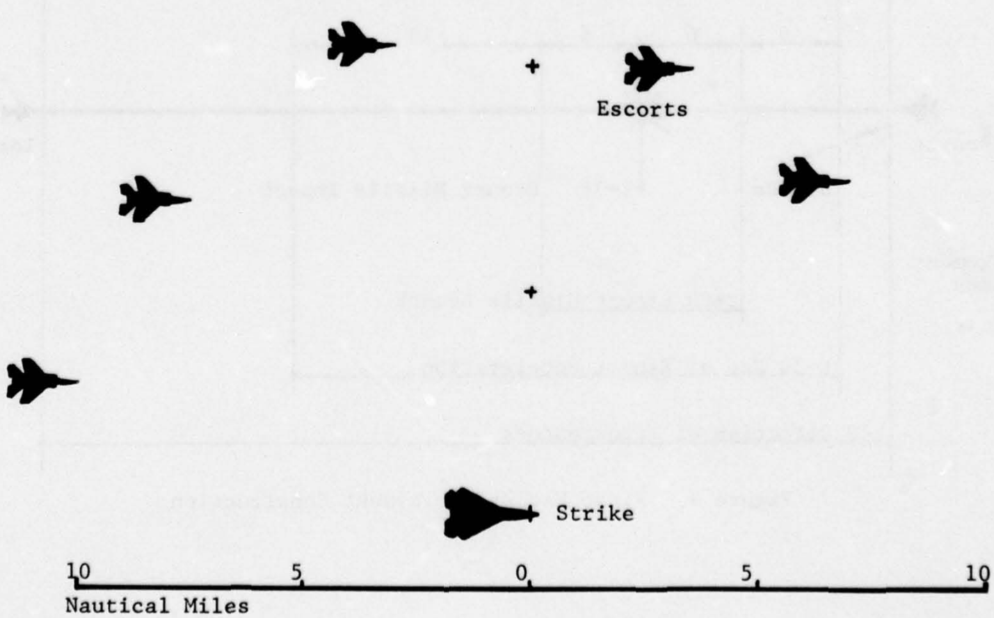

Figure 12; Second Escort Deployment

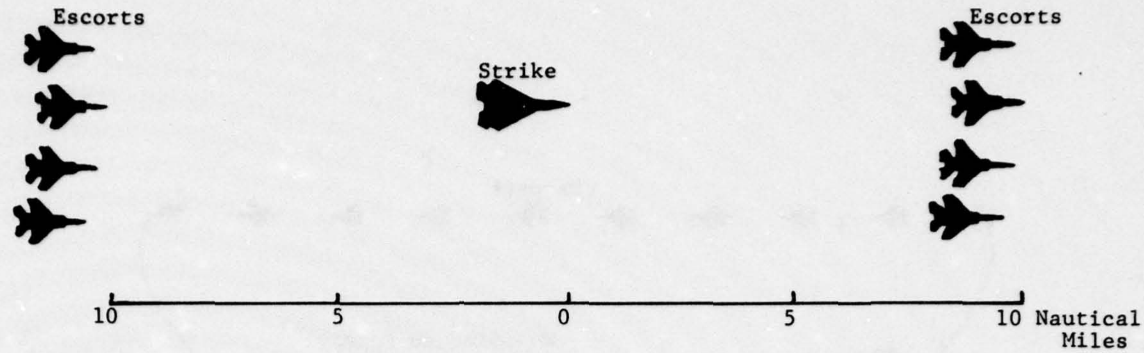

Figure 13; Third Escort Deployment

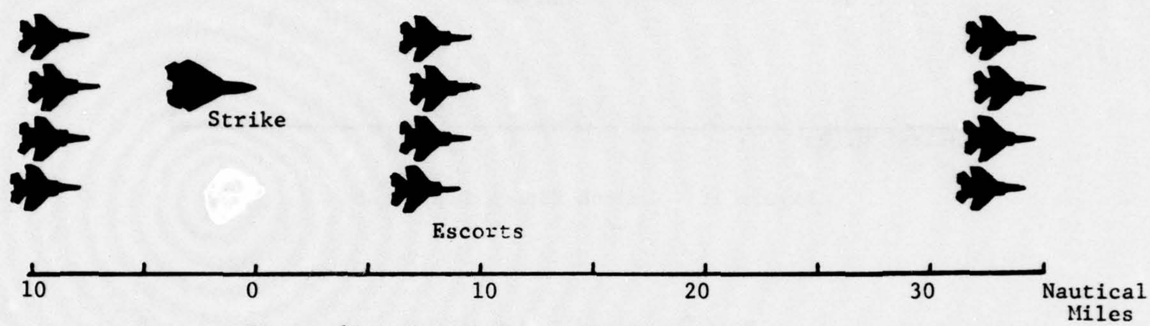

Figure 14; Fourth Escort Deployment

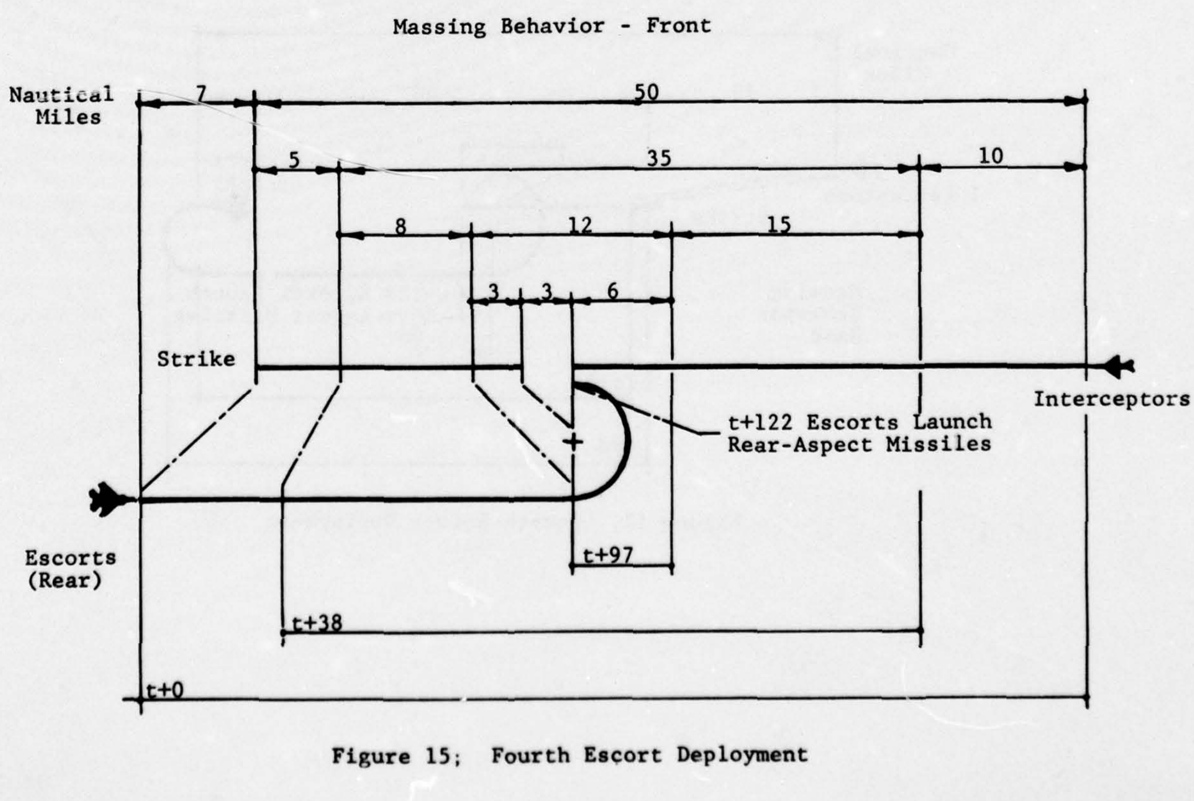

Figure 15; Fourth Escort Deployment

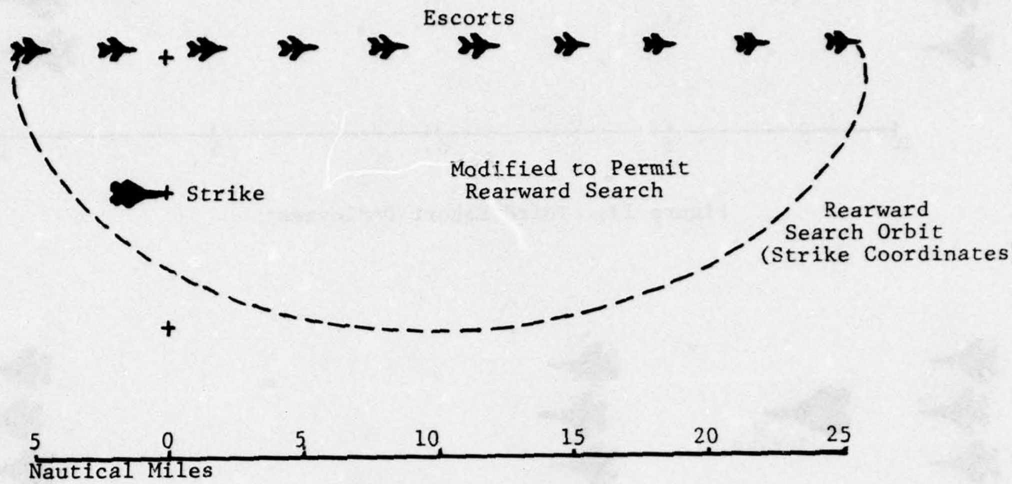

Figure 16; Fourth Escort Deployment

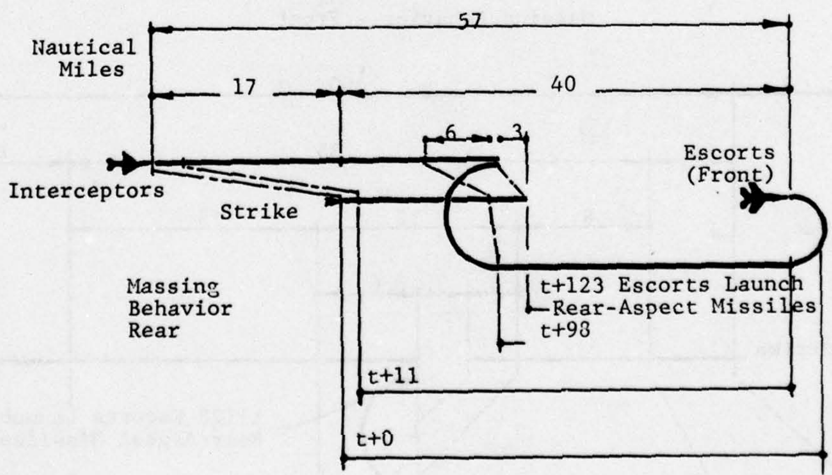

Figure 17; Fourth Escort Deployment

ADVANCED CONTROL CONCEPTS FOR FUTURE FIGHTER AIRCRAFT

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SUMMARY

The now available new techniques for the design of the control and stabilization systems lead to a remarkable improvement of the expected mission effectiveness and to the innovation of new modes of operation. The paper discusses the consequences of this new technology for the aircraft design and system lay-out mainly from a designer point of view.

In detail the possibilities and limits of these new control concepts as ACT and RSS and the criteria for the corresponding control surface lay-out are discussed. Furthermore requirements and the technical realization methods for a modern fly-by-wire digital control technology are presented. Finally some comments on current research programs and future prospects are given.

1. INTRODUCTION

According to the headline of the meeting the aspects of modern control concepts for future fighter aircraft design will be discussed in this paper.

Advanced Control Concepts means Active Control Technology (ACT), Reduced Static Stability (RSS) and Post-stall and Spin prevention combined with digital fly-by-wire technology (FBW).
Though the benefits of these new concepts are well-known since several years only now the necessary integrity and reliability of the electronic processors have come to the point at which by incorporating a reasonable failure management acceptable total system integrity values can be achieved, which are comparable to conventional control systems.

These so called "multi-mode control systems" lead to a remarkable extension of the possibilities of modern Control Concepts. Improved flying qualities with the possibility of an individual optimization according to particular operational tasks and new modes of operation lead to higher accuracy of the weapon delivery and to a superior maneuverability with reduced pilot work load.

The digital fly-by-wire technique opens a large amount of flexibility and selftest capability, which also influences the individual aircraft design and the design process.

2. ADVANCED CONTROL CONCEPTS AND THEIR POSSIBILITIES

Detailed description of the different types of the new systems and their possibilities have been given on the AGARD-Specialists-Meetings [1], in Advisory Report [3] and Lecture Series [2] so it is assumed, that the main aspects are well-known. Therefore it seems to be sufficient to give only a short summary of the main features. Those interested in more detailed information are referred to reference [2], where a good overview and an excellent bibliography on all aspects of modern flight control systems is given.

2.1 New degrees of freedom for control

In the past only the three rotatory degrees of freedom have been used for the primary control of the aircraft, fig. 1, whereas the translational degrees of freedom have been indirectly controlled by the rotatory, too. The direct force control technology (direct lift-, sideforce- and drag control) add new control inputs and therefore new degrees of freedom for the primary control.

New modes of operation

These new degrees of freedom for control lead in combination with modern multimode processors to the innovation of new modes of operation. Fig. 2 shows the new longitudinal modes.

Direct lift control:

Vertical flight path control at constant angle of attack which is a useful mode for pitch attitude corrections and minimizes the altitude loss during recovery.

Longitudinal Fuselage Pointing:

Pitch attitude control at constant flight path angle, which opens new possibilities for fire/flight control system integration.

Vertical Translation:

Vertical velocity control at constant pitch attitude, which is ideal for small vertical position changes such as formation flying or glide path adjustments.

Maneuver Enhancement:

a blended mode of DLC and conventional pitch control to reduce the error between pilot "g" command and aircraft response. A desirable fall-out of this mode is a gust alleviation mode if no "g" is commanded and the system reduces the "g's" of the gusts, which leads to an improvement of the ride control of the aircraft.

Fig. 3 shows the new directional modes:

Direct Sideforce:

directional flight path control by wings level flat turns, which is used for heading corrections and late lineup corrections avoiding sight pendulum effects.

Directional Fuselage Pointing:

directional attitude control at constant flight path angle, which corresponds to the longitudinal fuselage pointing mode.

Lateral Translation:

side velocity control at constant heading.

This mode is useful for small lateral position corrections in formation flight and the ideal mode for cross wind corrections.

2.2 Human Factors

In close connection to these modes of operation the new special cockpit lay-outs have to be seen with sidestick, keyboards and displays.

Experiences with these new modes of operation are available by the USAF/General Dynamics YF-16 CCV Fighter Program, more details in references [7], [8], [12], [13].

2.3 Optimization of Handling Qualities

This is another important aspect of the new technology as it is possible to optimize the pilot-in-the-loop Handling Qualities according to particular operational tasks by maneuver enhancement, blended control (aileron-rudder interconnect) and special values of system behaviour. Disturbance and command response of the system can nearly be optimized independantly. These types of systems are called "response command augmentation systems".

2.4 Performance improvements by RSS

By reduced static stability a significant improvement in lift, drag and weight can be achieved. Fig. 4 shows the results of a longitudinal RSS in comparison to a conventional lay-out. The improvements in lift and drag are depending on the lift coefficient and Mach number and reach values of about 10 % in the subsonic and 15 % in the supersonic flight regime at a subsonic instability margin of 10 %.

The possible drag reduction by a RSS-layout for the vertical tail is depending on the configuration but is normally small. In this case it is mainly the possible weight reduction by the smaller vertical fins which has to be considered.

2.5 Enlargement of the operational flight envelope

This can be done by the maneuver load control and by stall and spin prevention systems.
All these possibilities lead to an increased weapon delivery accuracy as has been shown by simulator studies, reference [2], fig. 5. These Vectored Lift Fighter Simulator Results of McAir show a significant improvement especially for the pilot controlled tasks.

3. DESIGN RECOMMENDATIONS

To get the full benefit of the Advanced Control Concepts special recommendations have to be regarded already in the early state of a new fighter aircraft design. These recommendations can be divided into two parts, one concerning the aircraft shape that means especially stabilizer and control surfaces and the other the systems.

3.1 Stabilizer and Control Surfaces

The design of the stabilizer and control surfaces is no more mainly influenced by stability criteria but has to be done according to trim and control requirements, which on their part can be influenced by the weapon and target dynamics.

H o r i z o n t a l   t a i l

In consideration of the possible reduced static stability (RSS) the horizontal tail area can be defined in combination with a proper c.g. position to get a minimum possible total drag. The principle is to have a lift carrying tail and to distribute the necessary lift on wing and horizontal tail in an optimal manner for a given design point.

Fig. 6 shows the general relation of tail area and c.g. position on the total drag for a given lift coefficient. The upper part of Fig. 7 shows the attainable minimum total drag as a function of the tail surface area, whereas the lower part shows the corresponding optimal c.g. position.
For these parametric calculations two philosophies for constant aspect ratios of wing and tail have been considered:

a. Starting at a given horizontal tail surface ratio the sum of the wing and tail area remains constant. That means increased wing area at reduced tail area and vice versa.

b. Starting at the same lay-out point the wing area remains constant. That means reduced sum of wing plus tail area at reduced tail area ratio.

In case (a), the minimum drag is realized by the smallest by control purposes possible horizontal tail. In case (b) there is existing a minimum of total drag, too, but at relative large tail areas. It is obvious that a new design will be done according to philosophy (a). Case (b) is valid for a supplementary modification of a given aircraft into a RSS-configuration. As the main reason of the good performance of a RSS-design is due to the carrying tail, it is obvious to try getting this effect also by a positive C_{Mo} of the wing-fuselage combination, that means a nose-up moment for zero lift. The third trace on the upper figure shows that the effect is of the same order as for the RSS. The additional benefit of this positive C_{Mo} can be seen from the lower part of the figure, where the optimal c.g. position and the limit for static stability or instability are presented. For usual tail areas the optimal c.g. position in the case of the positive C_{Mo} is in the stable region. That means no trouble with additional redundancy.

Of course it has to be confessed that it is not easy to get positive C_{mo} and there will be additional drag due to such a design by maneuvring flaps, cambered strakes or special wing profiles, which has not been taken into account here. But the intention of this graph is to put your attention on this interesting aspect and there are configurations possible, which have that additional benefit of a positive C_{mo} without the disadvantages of additional drag due to this fact.

Fig. 8 shows an example of a positive C_{Mo}-design. This is an artist's impression of a Dornier proposal of a non planar wing system (diamond wing) with the additional benefits of low induced drag at low Mach numbers due to the biplane-effect and reduced wave drag, at high Mach numbers due to the possible lower thickness of the wings.

For normal configurations - our calculations are based on a F-16 type design - it remains true, that negative static stability leads to the mimimum total drag. But the question is, what amount of negative static stability can be realized under consideration of control requirements.

Fig. 9 shows the tail lay-out diagrams in terms of tail area and the stability margin. Under the assumption that a c.g. range of 10 % mean aerodynamic chord will be necessary, it can be seen that though the tail area of the RSS lay-out is smaller as for the conventional case the most rear-ward c.g. position is given by control requirements at a stability margin of only -3 %. And from the landing gear geometry - if there is no c.g. change possible during the flight by fuel management - the most negative stability margin may not exceed -8 %. That means a RSS design requests a more rearward position of the main landing gear than usual. The lower part of the figure shows again the effect of a positive C_{Mo} and the influence of the control reserve.

In fig. 10 the effect of the non optimal design can be seen for the total drag. The drag penalty for the non optimal RSS configuration is low, as for the necessary tail volume of this configuration the drag-curve is relatively flat and the attainable stability margin is near to the optimum for this here chosen design condition. By RSS or by a positive C_{Mo} of 0.1 a drag improvement of about 15 % is possible.

For small C_L-values the difference diminishes. Finally it should be mentioned, that there exists another possibility to shift the optimal c.g. to more forward and therefore more stable positions by a positive

downwash angle a zero lift [16]. However our calculations have shown, that for higher C_L-values there are no drag improvements possible.

Vertical tail

The criteria for the vertical tail design are listed on fig. 11. Due to the relative high thrust to weight ratios of modern fighter design an engine failure is the design case for twin engine configuration, especially if the vertical tail is a fin plus rudder type. But even for an all movable fin an engine failure is dominant for the necessary vertical tail area.

For single engine configurations the vertical tail area is depending on dynamic or static stability requirements, where the case of the static weather cock stability $C_{N\beta}$ at $M > 1$ is determinative.
If the RSS philosophy can be used also for the lateral directional motion of the aircraft, because a fully redundant FBW-Concept for all control axis will be already available, then the requirements of the crosswind landing are valid for the vertical fin area. In this case a fin plus rudder configuration is the better solution in contrary to the engine failure case, where an all movable fin leads to a lower fin area. For some configurations with these vertical tail areas it is already possible to fulfill the dynamic stability speficiation in the case of an augmentation system failure at the reduced Level of 3 of the US-Handling-Qualities-Spec. MIL-F-8785B.

The question single or double vertical fin cannot be answered clearly. As can be seen on fig. 12, reference [4], [5], the designers of the F-16 and F-17 have come to opposite solutions by the same argument. One important reason for the individual solution seems to be the position of the strake-vortices on the rear part of the fuselage at high angles of attack. As a general guide-line it can be said that for a consequent CCV-configuration a single vertical tail will be sufficient, whereas for double engine configuration two vertical tails will be preferable.

Direct lift and side force devices

The design of these devices depends on the expected amount of additional lift or sideforce according to the intended tasks of the aircraft. The values will be higher for air superiority missions than for the improvements of the weapon delivery accuracy in air to ground tasks.

Fig. 13 shows the effectiveness of different possibilities to generate direct side forces as a function of the angle of attack. It can be seen that the usual configuration with vertical canards is not the best concerning the angle of attack relation. Very good results have been achieved by differential deflected horizontal canards [1] which produce the side force mainly indirectly by interference effects. In the same reference data of a corresponding configuration for the YF-16 are shown which are above the values of this diagram. In spite of this excellent efficiency no practical application is known so far.

The pylon split flaps are a special possibility proposed by Dornier which will be tested by an Alpha-Jet within a flight test program sponsored by the German Ministry of Defense. To give the non-aerodynamicist a feeling of the attainable performance with these devices fig. 14 shows the normal respectively side acceleration and the side slip angle as a function of the DLC or DSFC-coefficient for different flight conditions. Requirements of necessary values for the different tasks will be available as a result of the current flight test programs.

3.2 Control and Stabilization Systems

The recommendations for modern control and stabilization systems have become more detailed and severe concerning not only the stability and control of the aircraft but including the mission, weapon and target dynamics. Fig. 15 gives an overview of the total dynamic system, which has to be considered. Though due to the complexity of the problem, research and development efforts concentrate on individual elements, the success of the task depends on the performance of the total closed loop system.

The heart of the system which has to be considered here are the two "Multi Mode Control" (MMC) boxes which have been divided into two parts. The feed back branch is mainly responsible for adequate system dynamics whereas the forward branch can be characterized as command module.

Even for the feedback special task depending recommendations are necessary. For instance it is no more reasonable to have as much damping as desirable from the pilot point of view. Fig. 16, upper part, shows the influence of increased pitch damping on pilot opinion and the performance for an air-to-ground bombing mission. Though the pilot rating leads to better judgements with increasing Mq the performance became worse, as this task requires precise flight path or velocity vector control, which can be better done by an aircraft with reduced pitch damping. The contrary has been found for air to ground gunnery, which requires precise attitude control and therefore a high pitch damping.

In general it can be precised that system characteristics should be optimised according to the individual task as it is illustrated in the lower part of fig. 16.

The root locus presentation is of course only valid for linear system dynamics. As an example including nonlinear systems combined with a more closed loop consideration of the dynamic behaviour the C*-Criterium is used as longitudinal transient motion recommendation. The parameter C* is defined by

$$C^* = N_{Z_{pilot}} + K\,\Theta$$

and is influenced by the pitch rate Θ at low speeds and by normal acceleration at the pilot's station at high speeds, Fig. 17.

As a special aspect of the modern control and stabilization systems the stall and spin prevention equipments have to be regarded. For these systems no general rules can be defined as the maneuver limitations are very strongly depending on the individual configuration. In fig. 18 the maneuver limitations of the Harrier and the A-7 aircraft, ref. [14], [15] are compared. In the case of the Harrier an unstable "roll off" is the maneuver limitation and for the A7 it is an unstable yaw divergence depending additionally on the side slip angle. The flight mechanical background and the significant parameters of the departure behaviour are not yet fully understood. It is now the task of an AGARD-FMP Working Group to find more general relationships and criteria.

3.3 Fly-by-wire digital technology

The fly-by-wire digital technology is essential for Advanced Control Concepts. In spite of the fact, that there are still some problems to solve, the advantages of this technique are evident:

- good adaptability for changing a/c parameters
- hardware definition possible before software is finished
- low cost of modifications during the development phase
- higher level of integration attainable (multiplex operation)
  easy realization of multivariable control
- good self test capability
- easy realization of nonlinearities

- no drift
- possible extension for additional tasks.

The main problem which has to be solved for this technology was the reliability. To day the reliability of these systems corresponds to mechanic/hydraulic control systems whereby the technical effort could be reduced by an advanced technology. The requirement for the total control systems reliability remains at the level of a mean time between failure (MTBF) equal to MTBF $\sim 10^7$.

That means a higher reliability for the subsystems and leads to the question: How many errors have to be survived and what happens in the case of a failure. This is specified by an "Operational Requirement", which today represents the philosophy of two equal error covering (fail-op)^2.

Fig. 19 shows the to-day approach and alternative modifications which would lead to further system simplification and should be regarded as a discussion basis. The to-day approach of totally covering two equal errors (fail-operate/fail-operate principle) led to quadru-redundant systems as realized at the F16 and the CCV-F104. The failure identification is done by majority decisions, which is a well-known technology but expensive.

The other two possibilities allow a degradation of the system performance in the case of a second error. At the left principle the degradation occurs only at the second error, but the pilot has no indication of the first error and will continue his task. At the right principle there is a two step degradation but the pilot will receive an indication of the first error and abort the mission.

These proposed modifications of the Operational Requirements together with the improved reliability of today electronical circuits (micro processors) 30 ÷ 50 % hardware can be saved for the same task and still improving the total system performance. The technical realization of the reliability requirements are done with the aid of the following techniques:

- redundancy
- failure self-monitoring
- software checks
- sensor skewing
- control surface redundancy
- dissimilary redundancy

Within these possibilities the failure self monitoring plays a major role and represents the most significant progress in this field. There are different methods of this technique:

- wrap around technique
- double processor technique
- watch dog timing
- model technique
- parity check/self correcting codes
- memory protection

"Wrap around" means the computer votes about his own outputs to the signal user to use or to omit the signal. Watch dog timing means that the omission of a periodical demanded test signal indicates an error. The model technique is especially used for the actuator failure self detecting.

Another aid of the redundancy management was called "sensor skewing". Fig. 2o gives an example of airflow direction sensors used at the CCV-F104:

Instead of 4 angle of attack and 4 sideslip sensors, a skewed sensor system with only a total of four sensors has been installed. Each sensor measures a combination of α and β and already two sensors are sufficient for both angles, that means the arrangement is fail-op, fail-op.

The similar principle can be used for control surface redundancy:

Double functions of already available control surfaces but no additional.
The dissimilary redundancy (simple back-up systems) is not yet realized and still disputed.

Of course these new techniques have also some problems and dangers by generic errors, programmers mistakes or production failures, which have to be solved by increased tests and improved check-out methods and procedures.

4. CURRENT RESEARCH PROGRAMS

The US programs are well-known. The results have been partly presented at AGARD's or other meetings. The most important are the

- NASA F-8C digital fly by wire research program
- General Dynamics/USAF YF-16 CCV Fighter program

On the more theoretical field McDonnell/Douglas has presented an example of an advanced fighter design based on today knowledge of advanced control concepts by its VLF (Vectored Lift Fighter), the development of which was based on a lot of simulator studies within the AFTI-Program [17].

In Germany two programs can be looked as relevant for this new technique, which both are sponsored by the German Ministry of Defense:

- MBB CCV F104-G by which up to 20 % MAC longitudinal instability should be tested (development and flight testing of a quadruply redundant fully digital guidance and control system) Fig. 21

- Dornier Alpha-Jet DSFC (development and flight testing of a simple direct side force concept by pylon split flaps) Fig. 22.

5. LIMITS AND FUTURE PROSPECTS

Though the new systems are very potential, they cannot correct all deficiencies of the bare airframe. Especially the stall and spin behaviour and the maneuver limitations can only be influenced up to a limited level. Therefore more exact knowledge of these phenomena has to be available.

New lay-out criteria for new handling modes and control devices (side stick) are necessary. The old criteria for second order systems and center stick controller are no more valid.

The development of multi loop pilot models for the first optimization seems to be desirable in combination with new accommodated simulation techniques on ground and in flight. In combination to that some human factors effort will be necessary to clear the possibility of the pilot to control all these new modes without increased pilot workload.

Generally it can be concluded that in future Advanced Control will mean an optimal integration of displays, flight mechanics of the basic airframe, system and weapon dynamics and the pilot behaviour to get the highest possible performance for each individual task of a fighter aircraft mission.

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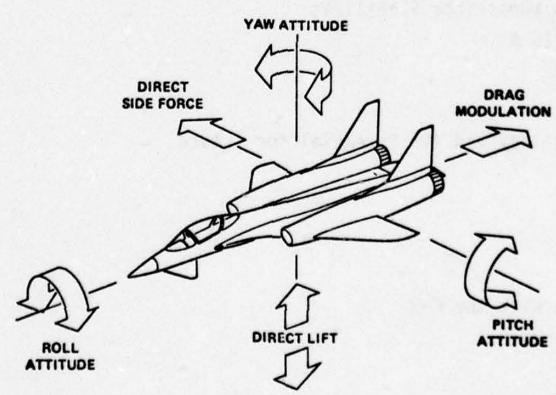
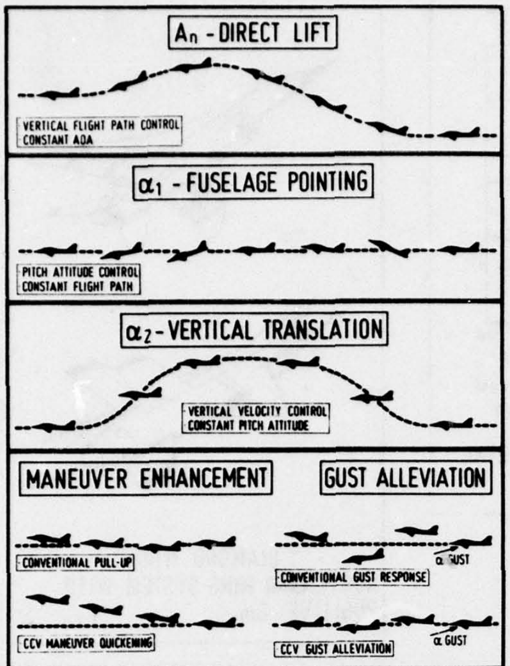


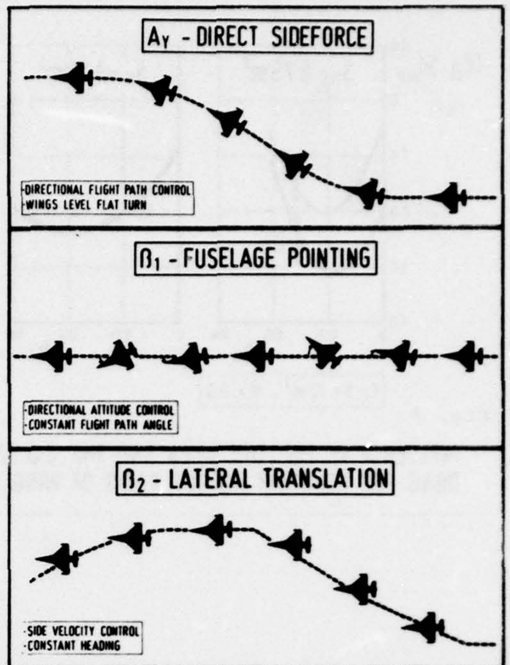
Fig. 1
CONTROL IN SIX DEGREES OF FREEDOM



NEW LONGITUDINAL MODES OF OPERATION

Fig. 2

REFERENCE 12



NEW DIRECTIONAL MODES OF OPERATION

Fig. 3

REFERENCE 12

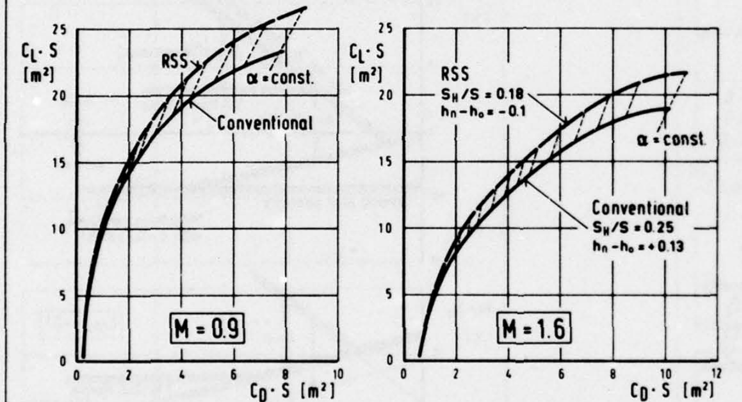


Fig. 4

LIFT AND DRAG IMPROVEMENTS DUE TO RSS

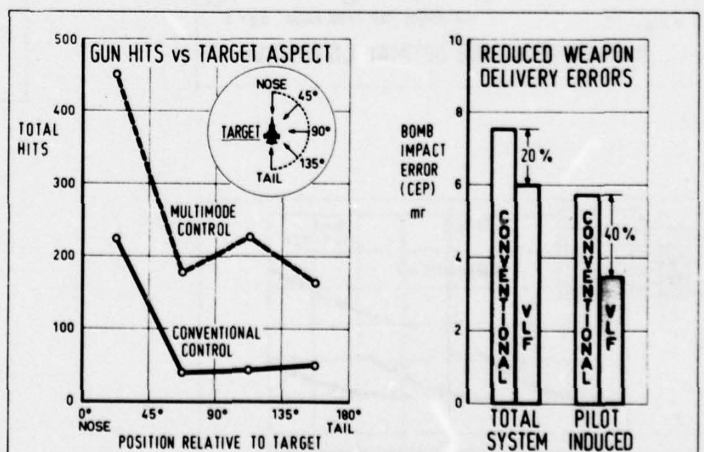


Fig. 5

VECTORED LIFT FIGHTER SIMULATOR RESULTS

REFERENCE 2

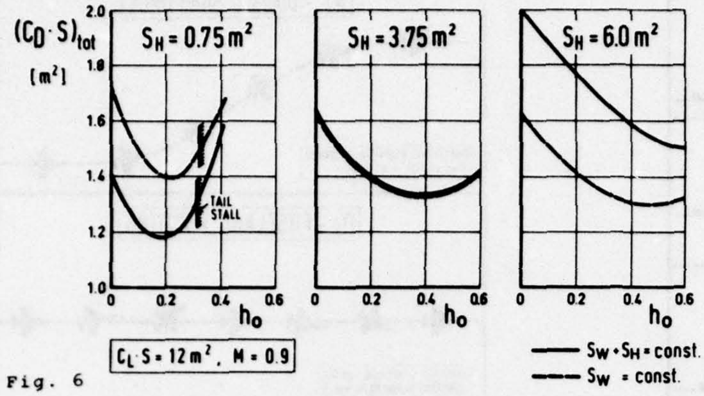
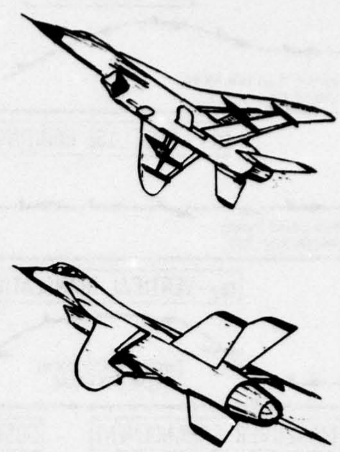


Fig. 6

INFLUENCE OF THE TAIL AREA AND THE C.G. POSITION ON THE TOTAL DRAG FOR CONSTANT ASPECT RATIO OF WING AND TAIL



" DIAMOND WING"
NON PLANAR WING SYSTEM WITH POSITIVE Cm_o

Fig. 8

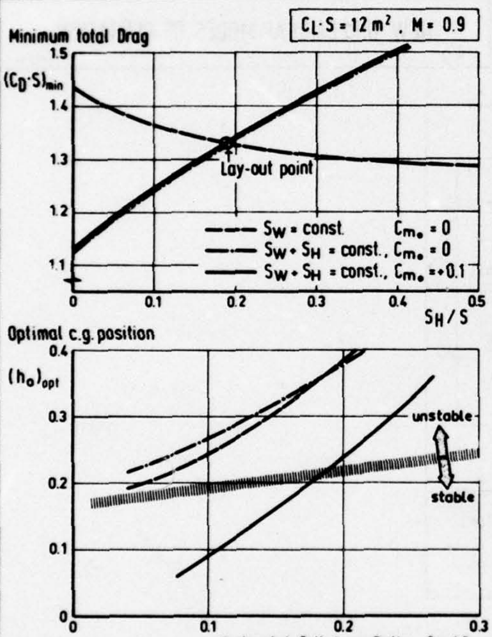


Fig. 7

MINIMUM DRAG AND OPTIMAL C.G. POSITION

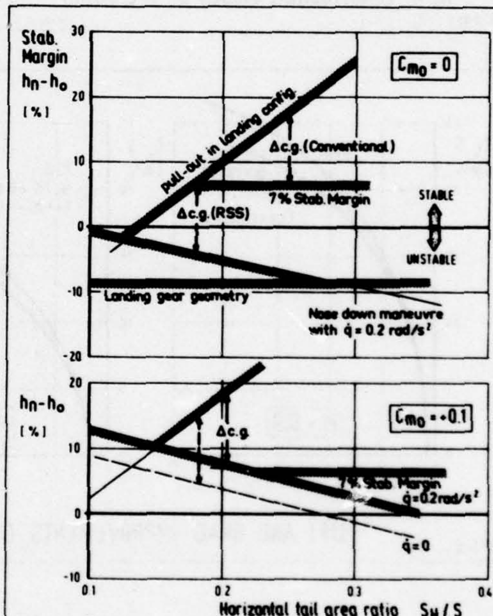


Fig. 9

HORIZONTAL TAIL AREA LAY-OUT

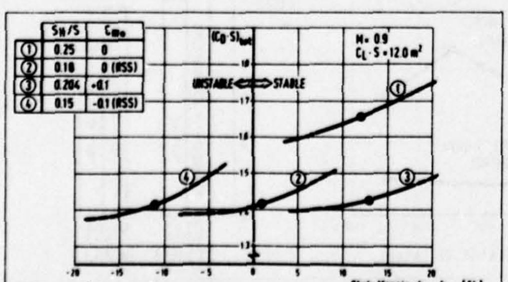


Fig. 10

TOTAL DRAG OF REALIZABLE HORIZONTAL TAIL - C.G.-COMBINATIONS

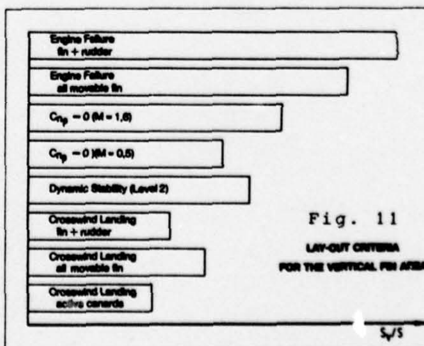


Fig. 11

LAY-OUT CRITERIA FOR THE VERTICAL FIN AREA

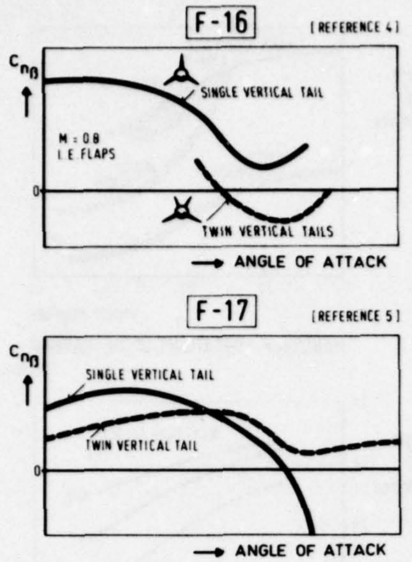


Fig. 12

LAY-OUT CRITERIA FOR SINGLE OR TWIN VERTICAL TAIL

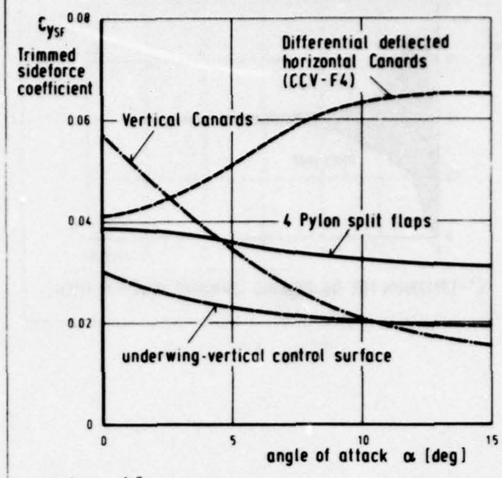
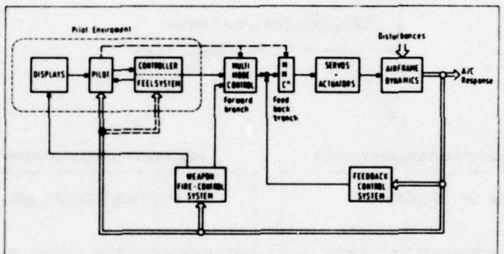


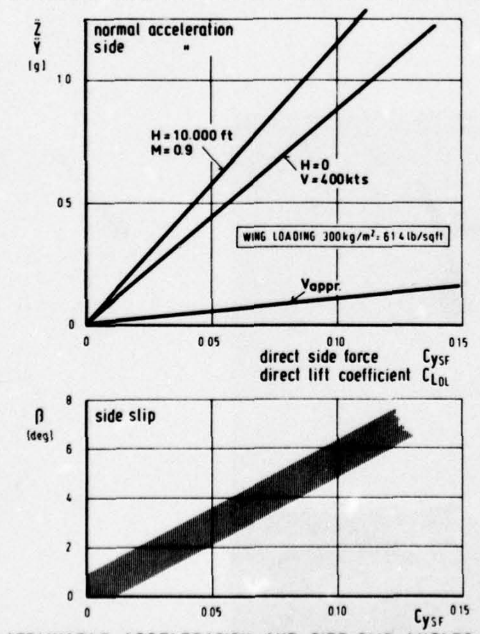
Fig. 13

THE EFFECTIVENESS OF DIFFERENT METHODS TO GENERATE DIRECT SIDE FORCES



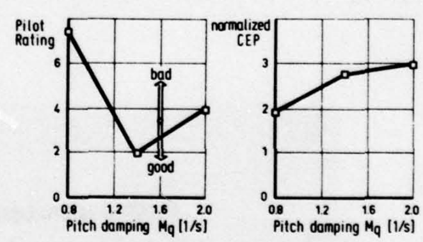
DYNAMIC SYSTEM OVERVIEW

Fig. 15



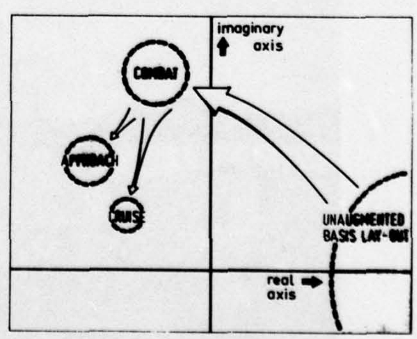
ATTAINABLE ACCELERATION AND SIDE SLIP ANGLES AS A FUNCTION OF THE DIRECT FORCE COEFFICIENTS

Fig. 14



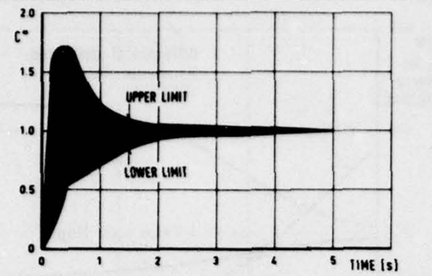
INFLUENCE OF PITCH DAMPING ON CEP AT "AIR TO GROUND BOMBING"-MISSION

REFERENCE 3



OPTIMAL LONGITUDINAL ROOT LOCUS-POSITIONS FOR DIFFERENT MISSION PARTS

Fig. 16



C*-CRITERIUM FOR THE DYNAMIC TRANSIENT MOTION IN PITCH

Fig. 17

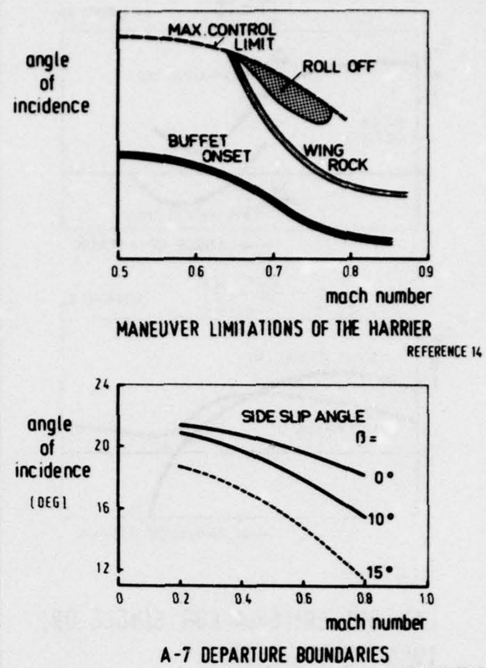
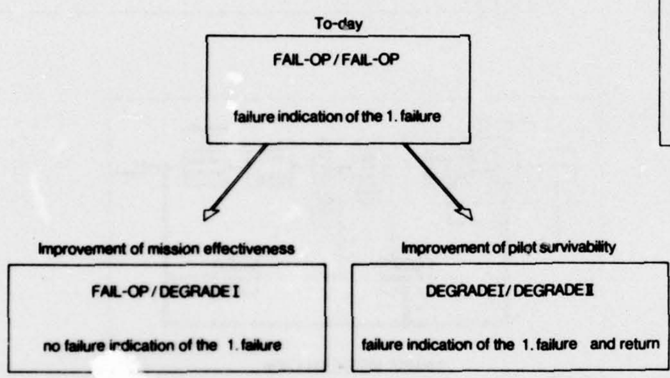


Fig. 18

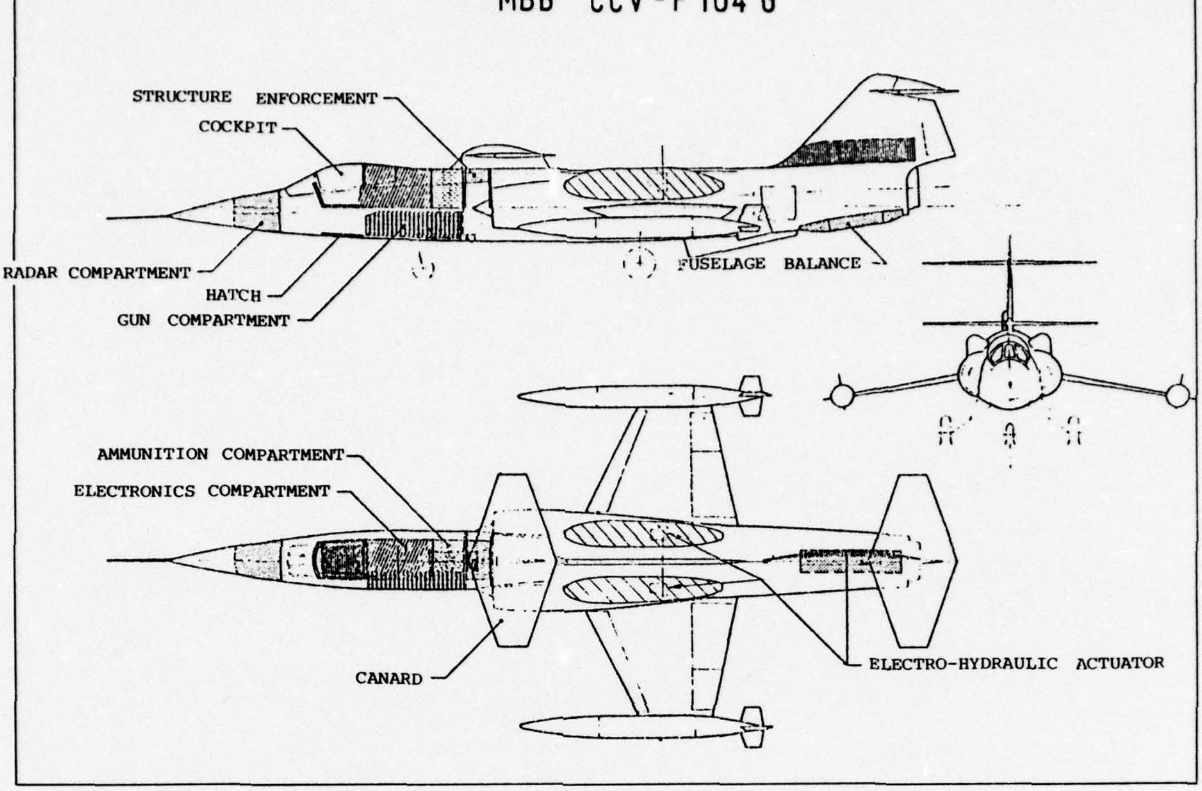


POSSIBLE MODIFICATIONS OF THE OPERATIONAL REQUIREMENTS FOR REDUNDANCY MANAGEMENT

Fig. 19

Fig. 20 SENSOR SKEWING

Fig. 21


MBB CCV-F 104 G
STRUCTURE ENFORCEMENT
COCKPIT
RADAR COMPARTMENT
HATCH
GUN COMPARTMENT
FUSELAGE BALANCE
AMMUNITION COMPARTMENT
ELECTRONICS COMPARTMENT
CANARD
ELECTRO-HYDRAULIC ACTUATOR

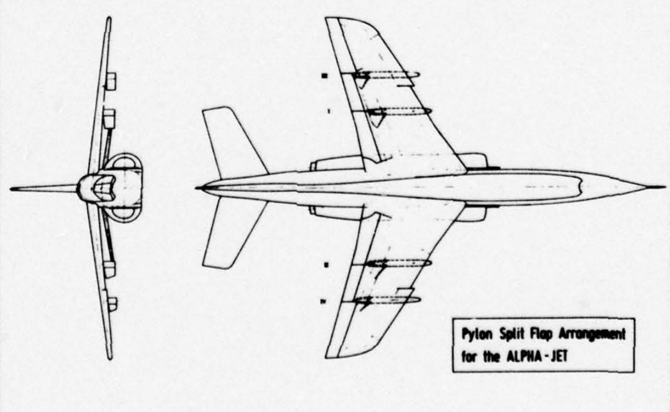

Pylon Split Flap Arrangement
for the ALPHA-JET

Fig. 22

FIGHTER SUPERIORITY BY DESIGN

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Summary

Free-world defense budgets will not permit numerical equality with potential adversaries in the foreseeable future, and will place bounds on the allowable costs of weapon systems. Fighter aircraft design teams must seek force multipliers - factors that permit one fighter to be the equivalent of many less effective fighters. Combat performance, firepower and weapon system capability are such multipliers, but cost constraints prevent looking only to combat capability for the required multiplying effects. Multimission versatility permits satisfaction of time-varying mission requirements with smaller total forces, while providing economies in development and production-quantity reductions in unit costs. Increases in reliability and maintainability contribute to improved availability - a force multiplier - while simultaneously reducing operations and maintenance costs. Increased combat survivability, by prolonging combat life, multiplies combat effectiveness and decreases combat costs, often with concomitant peacetime safety cost benefits. Reconciliation of such force multipliers with one another and with peacetime costs is illustrated by selected aspects of the U.S. Navy/U.S. Marine Corps F-18 program.

Introduction

Our national budgets are strained by popular demands for an increasing variety of governmental services. Neither defense budgets nor conscription policies of the free nations permit us to match the potential adversary man-for-man, tank-for-tank, and aircraft-for-aircraft in peacetime (Figure 1). We must expect to be outnumbered in combat. A fast-breaking war would permit little augmentation of our forces in being. We must compensate for inferior numbers by seeking force multipliers - factors which will allow one fighter aircraft, for example, to perform combat functions that would require greater costs for larger numbers of lower-quality aircraft. The necessity of this approach has been emphasized over and again by history. This discussion will center upon illustrating, by current example, our continuing capability to provide the needed qualities in fighters within peacetime constraints.

FREE WORLD OUTNUMBERED

- BUDGET PRIORITIES
- CONSCRIPTION POLICIES

REQUIRE FORCE MULTIPLIERS

- SUPERIOR CAPABILITY
- MULTIMISSION VERSATILITY
- AVAILABILITY
- SURVIVABILITY

GP77-0979-2

FIGURE 1
PEACETIME IMPLICATIONS

Superior combat capability - performance and weapon system lethality - does provide a force multiplier, the ability to succeed against superior numbers. Peacetime cost constraints, however, prevent viewing greater and more costly unit capability as the sole measure of superiority, the only force multiplier. The incentives are very high for fighter aircraft designers to use their ingenuity to maintain superior combat abilities consistent with peacetime cost constraints, and concurrently contribute to public awareness of the need for superior abilities.

In addition to combat capability, this article will explore, as force multipliers, the effects of multi-role abilities, increased availability, and improved survivability while maintaining adequate levels of firepower and overall wartime lethality. These have a historical record as force multipliers in successful tactical aircraft, including the F-4 Phantom and F-15 Eagle used here for perspective. Aspects of the U.S. Navy/Marine Corps F-18 Hornet fighter program will be included herein for currency in illustration.

The need to balance contributors to wartime effectiveness against peacetime constraints (Figure 2) will receive due attention, the peacetime emphasis dealing with minimization of total life cycle costs. Fortunately, peacetime cost minimization and combat effectiveness maximization do not always point toward different design solutions. Many design options contribute to both objectives. Where they do not we must continue to seek minimum-cost solutions to fighter requirements, but we must assure that they are true solutions and not merely minimum-cost designs. In addition to being affordable, fighters which will be effective in wartime must provide lethal firepower in unpredictable situations. It has been often observed, and more often forgotten, that we "never fight the war we design for," and often use our aircraft most effectively in ways not originally intended. Good designs are adaptable.

Fighter design emphasis has varied through the years with notable errors as well as successes. But an important continuity can be illustrated by examining the original philosophy of two current operational fighters, the F-4 Phantom and the F-15 Eagle, and the F-18 Hornet now in development. These aircraft already span a 23 year period, but over one-half century will undoubtedly be spanned by these aircraft before they all leave operational forces, a rather important illustration of the adaptability possible in fighter design even in radically changing technical, political, and threat environments. A chronology, Figure 3, provides perspective as well as emphasizing the very long lifetime expected for multirole capable designs.

As political historians differ in their perceptions and accounts of the same events, so do participants and observers of aircraft programs vary in their views of the origins and character of the design. Thus, while the Design Philosophy presented herein in Figure 4 for the Phantom, Eagle and Hornet is generally accepted, there is room for some debate (particularly because of its shortened form).

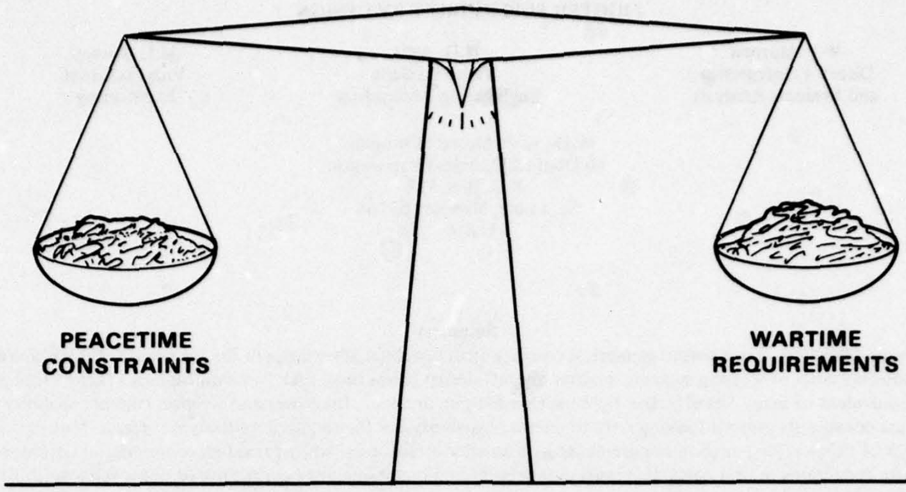

FIGURE 2
BALANCE IS ESSENTIAL

	F-4 PHANTOM	F-15 EAGLE	F-18 HORNET
DESIGN CONTRACT START	JUN 1955	JAN 1970	JAN 1976
FIRST FLIGHT	MAY 1958	JUL 1973	SCHEDULED FALL 1978
PRODUCTION RUN START	1960	1974	1980
BEGIN OPERATIONAL SERVICE	1961	1976	1982
EXPECTED PRODUCTION COMPLETION	BEYOND 1979	BEYOND 1985	BEYOND 1990

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FIGURE 3
AIR SUPERIORITY FIGHTER CHRONOLOGY

F-4 PHANTOM VINTAGE 1960

1. PERFORMANCE
   – SPEED
   – ACCELERATION
2. LETHALITY
   – ALL-WEATHER
   – LARGE PAYLOAD
3. CARRIER SUITABILITY
4. ADAPTABILITY
   – FIGHTER
   – ATTACK
   – RECONNAISSANCE
   – SPECIAL MISSION

F-15 EAGLE VINTAGE 1975

1. LETHALITY
   – MULTITHREAT
   – ALL-WEATHER
   – ELECTRONIC WARFARE
2. PERFORMANCE
   – MANEUVERABILITY
   – ACCELERATION
   – SPEED
3. SURVIVABILITY
4. AVAILABILITY
   – HIGH RELIABILITY
5. LONG SERVICE LIFE
6. ADAPTABILITY

F-18 HORNET VINTAGE 1981

1. MULTIROLE (99% COMMON)
   – FIGHTER
   – LIGHT ATTACK
2. LETHALITY
   – ALL-WEATHER
   – ELECTRONIC ENVIRONMENT
3. AVAILABILITY
   – HIGH RELIABILITY
   – LOW MAINTENANCE
4. CARRIER SUITABILITY
5. PERFORMANCE
   – MANEUVERABILITY
   – ACCELERATION
6. AFFORDABILITY
   – LOW INITIAL INVESTMENT
   – LOW OPERATING COST
7. LONG SERVICE LIFE

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FIGURE 4
FIGHTER DESIGN - PHILOSOPHY AND EMPHASIS

It is helpful in understanding their design philosophy to know the design mission for these aircraft. The Phantom was initiated by Letter of Intent in December 1954 for a single place, attack aircraft designated the AH-1. This mission was changed in June 1955 to Fleet Air Defense Interceptor with missile armament (all-weather and infrared), a radar (system) operator added, and the aircraft redesignated the F-4H. The Eagle's design mission was Air Superiority. The Hornet is a multirole design, all-weather air superiority and light attack, with both versions required to be virtually identical.

It is pertinent that the design approach to both the Phantom and the Eagle has been properly characterized as "not one pound for air-to-ground." History already records the major air-to-surface contributions by the Phantom after initial purchase as a Fleet Interceptor. There is no technical reason to expect different from the Eagle, although the growing needs for its unique qualities in the crucial air superiority role may delay exploitation of high air-to-surface capabilities inherent in the current design. The point is often missed that shrewd design can capitalize on qualities inherent to good fighters, allowing adaptation to new roles without significant initial compromise to the principal mission. If those qualities fundamental to good fighters are shortchanged in the initial design, then not only will subsequent multi-role effectiveness be inherently inadequate, but useful life as a combat aircraft in its initial mission will be unnecessarily brief, at very great expense to the user.

The background and circumstances that led to the variations in Design Philosophies and their relative emphases would require great length to recount. Major contributing factors have been:

a) Technology Advances (such as the advent of missiles, structural material advances, solid state and digital electronics, high thrust-to weight propulsion, aerodynamic configuration advances, advances in engineering methods, and system design capabilities).

b) Threat Changes (including changes in quality, quantity, character and emphasis).

c) Actual Wartime Experience (including Korea, Southeast Asia, mid-East).

d) Economics (including reduced defense budget share of the gross national product and rapid rise in personnel cost).

The increase in the number of items demanding emphasis is very real and properly denotes an increasingly complex design challenge for each new generation of aircraft. But some items are seen to recur as principal design factors, through with different relative emphasis. These principal issues are lethality, performance, availability, and adaptability. Note that adaptability is primary, by definition, in a multirole design such as the Hornet, and that such adaptability is set forth in specifications because of the large cost savings this approach provides to the customer (the affordability concept). Some specifics of how this philosophy impacts design and force multiplier qualities will be subsequently discussed.

A brief description of the Hornet program is called for (Figure 5) to place this material in perspective. That program is intended to provide the U.S. Navy and Marine Corps with fighter replacements for their F-4 Phantoms, while additionally providing light attack aircraft as replacements for the Navy's A-7. The two F-18 fighter versions and one A-18 attack version have 99 percent commonality. The basic aircraft is designed for one-man operation; a two-place trainer is included in the planned production, and retains all of the combat capability of the basic fighter except as limited by a slightly reduced internal-fuel volume.

PURPOSE

- **REPLACE F-4 NAVY FIGHTER**
- **REPLACE A-7 NAVY ATTACK**
- **REPLACE F-4 MARINE FIGHTER/ATTACK**

PRODUCTION

- **800 AIRCRAFT**
  - **430 FIGHTERS**
  - **310 ATTACK**
  - **60 TRAINERS**

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FIGURE 5
F-18 PROGRAM

The F-18 (Figure 6) has two General Electric F-404 engines, a moderately swept wing, and a leading edge extension (LEX) that enhances both performance and handling qualities. Basic fighter armament includes two AIM-7F Advanced Sparrow missiles, carried conformally on the fuselage, two AIM-9L Sidewinders on the wing tips, and an internal 20 mm cannon. Five other stores stations permit carriage of up to three external fuel tanks, additional air-to-air missiles, or a variety of air-to-ground ordnance.

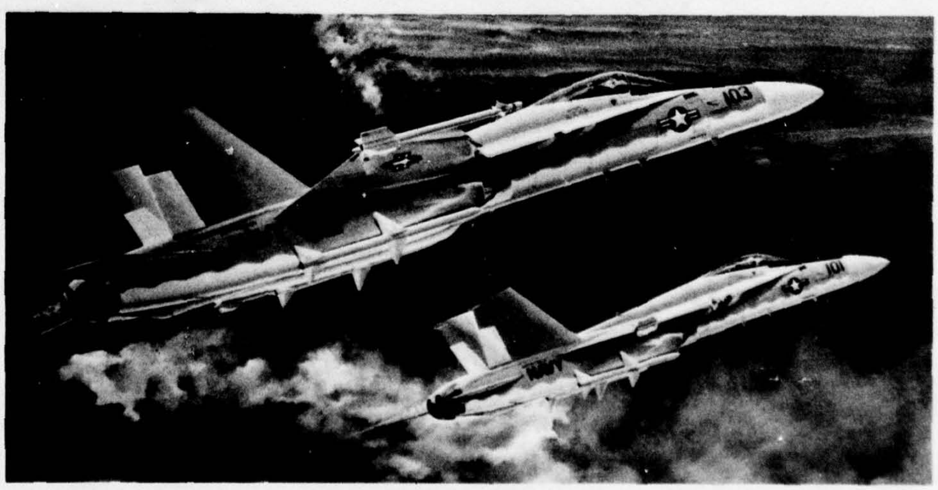

FIGURE 6
F-18 ARTIST'S CONCEPTION

Combat Capability as a Force Multiplier

Combat capability encompasses aerodynamic performance and weapon system capabilities. These traditional concerns of the designer will be treated briefly, in keeping with the emphasis on nontraditional contributors to combat effectiveness.

The Hornet leading edge extension (Figure 7) creates a vortex which generates a low pressure field on the upper surface of the LEX and wing as the vortex passes downstream over the inboard portion of the wing. Benefits of the LEX include an increase of more than 50 percent in maximum lift, reduction in drag due to lift and in supersonic trim drag, and reduction in buffet intensity. The moderately swept basic wing assures good flying qualities and departure resistance. The basic wing camber was selected to provide maximum supersonic turning performance with very little degradation in acceleration. Leading edge and trailing edge flaps are automatically positioned for subsonic and transonic maneuvering as a function of angle of attack and Mach number. These flaps also provide the high lift required for good carrier approach characteristics. Except for the carrier suitability characteristics, the basic aerodynamic concept has been demonstrated in the flight test program of the Northrop YF-17, from which the F-18 has been adapted.

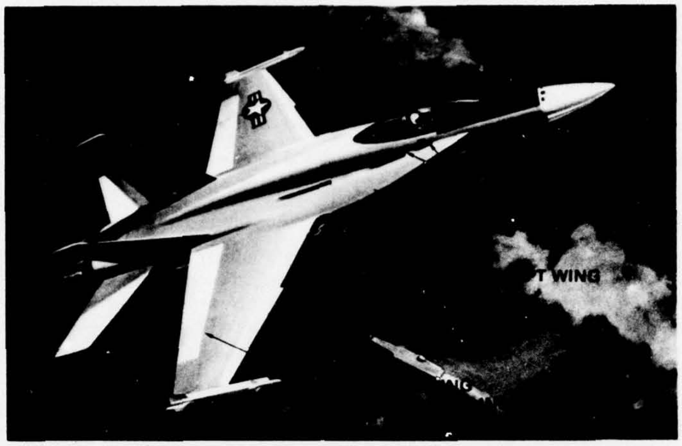

FIGURE 7
AERODYNAMIC FEATURES

Aluminum alloys dominate the structural material selection (Figure 8), providing lightweight, ease of fabrication, corrosion resistance, and low cost. A limited amount of titanium is used, primarily in hot sections. Graphite/epoxy composites are used extensively to save weight and cost where complexity, inspection, and repair access permit their consideration. Major composite applications are in the empennage, wing skins, and access doors.

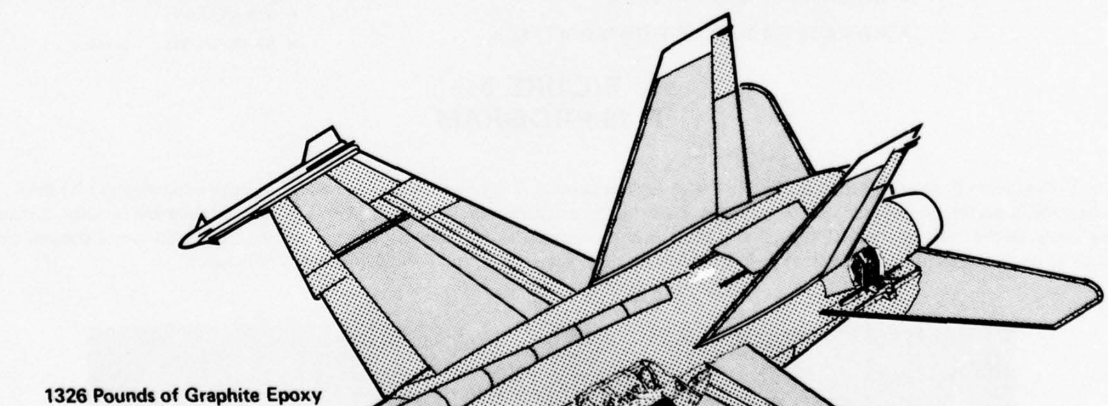

FIGURE 8
F-18 MATERIALS DISTRIBUTION

Advanced avionics technology permits significantly reduced subsystem weights. The Hornet radar, installed, is 600 pounds lighter than the F-4J radar, a reduction of almost one-half, even though it provides increased range and greater flexibility of modes. The F-404 engine has a 60 percent higher thrust-to-weight ratio than the J-79 engine in the F-4J, and has better specific fuel consumption. Both attributes of the F-404 contribute to decreased takeoff gross weight.

Fighter performance still has a high priority in design (Figure 9). An indicator of the relative performance of the F-18 and F-4J is provided by the 31 percent improvement in combat thrust-to-weight ratio and the 10 percent reduction in combat wing loading, which tend to provide superiority in acceleration and turning performance, respectively. Despite having little more than two-thirds of the F-4J's takeoff gross weight, the Hornet also outperforms the F-4J in mission radius with equivalent loadings (Figure 10).

	HORNET RELATIVE TO F-4J*
TOGW	0.7
FUEL	0.77
ENGINE THRUST (MAX)	0.9
COMBAT WEIGHT	0.7
COMBAT	
THRUST TO WEIGHT	1.3
WING LOADING	0.9

*Values Rounded

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FIGURE 9
WEIGHT-SENSITIVE PERFORMANCE INDICATORS

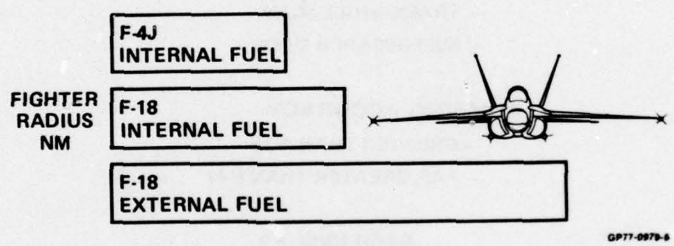

FIGURE 10
SUPERIOR MISSION PERFORMANCE

F-18 acceleration time (Figure 11) is 40 percent less than that of the F-4J. The maximum-power sustained load factor of the Hornet is about 50 percent greater than the F-4J's in the heart of the transonic maneuvering envelope (Figure 12). In intermediate power, the F-18 completely dominates the F-4J because of the lower augmentation ratio of the F-404 engine. That superiority gives the Hornet a much higher exchange ratio than the F-4J against maneuvering threats, as evaluated by digital air battle simulation.

Figure 13 shows that weapons system capability of the Hornet is superior to those of the F-4J and A-7, in both fighter and attack missions. Air-to-air radar range is more than double the F-4J's range in the demanding tail-chase look-down mode. Track-while-scan capability enhances the pilot's ability to attack multiple targets sequentially, and the supersearch mode permits automatic, head-up radar lock-on in maneuvering combat. Increased bombing accuracy decreases the number of sorties required to destroy a target, thus serving as an obvious force multiplier (Figure 14).

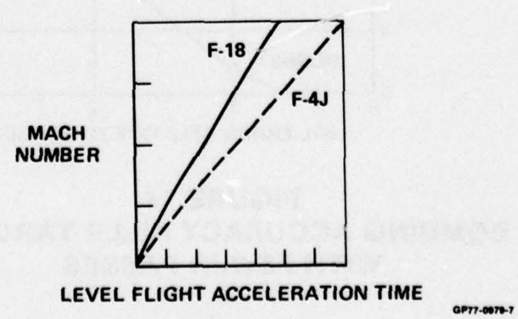

FIGURE 11
ENGAGEMENT CONTROL THROUGH ACCELERATION

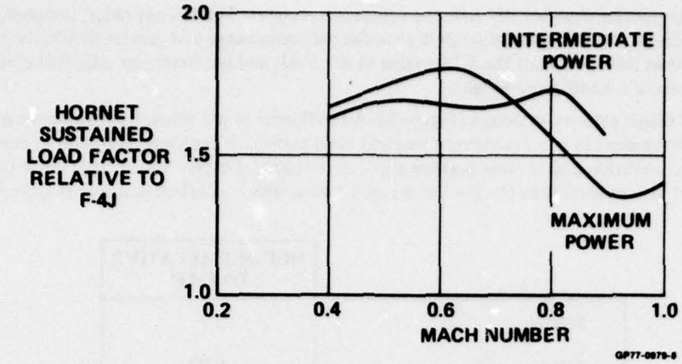

FIGURE 12
MANEUVERING PERFORMANCE IMPROVEMENT

RADAR (AIR-TO-AIR)

- DETECTION RANGE - LOOK DOWN
  - HEAD-ON F-4J EQUIVALENT
  - TAIL-CHASE OVER TWICE F-4J
- FEATURES NOT IN F-4J
  - TRACK-WHILE-SCAN
  - SUPERSEARCH MODE

BOMBING ACCURACY

- GREATER THAN A-7E
- FAR GREATER THAN F-4J

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FIGURE 13
HORNET WEAPON SYSTEM ADVANCEMENTS

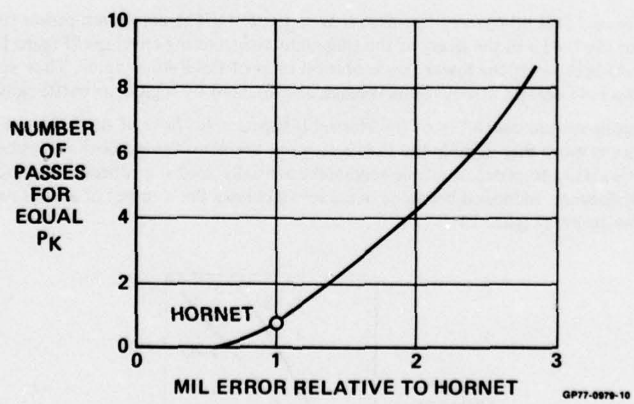

FIGURE 14
BOMBING ACCURACY KILLS TARGETS WITH FEWER PASSES

The Hornet concept emphasizes precision delivery for first-pass target kill with modest armament loads of either free-fall or guided weapons. In situations where maximum ordnance loads show benefits, however, (Figure 15) with suitable bomb racks the Hornet can carry bombloads comparing favorable with A-7 capability, with rough equivalence on a payload-radius basis.

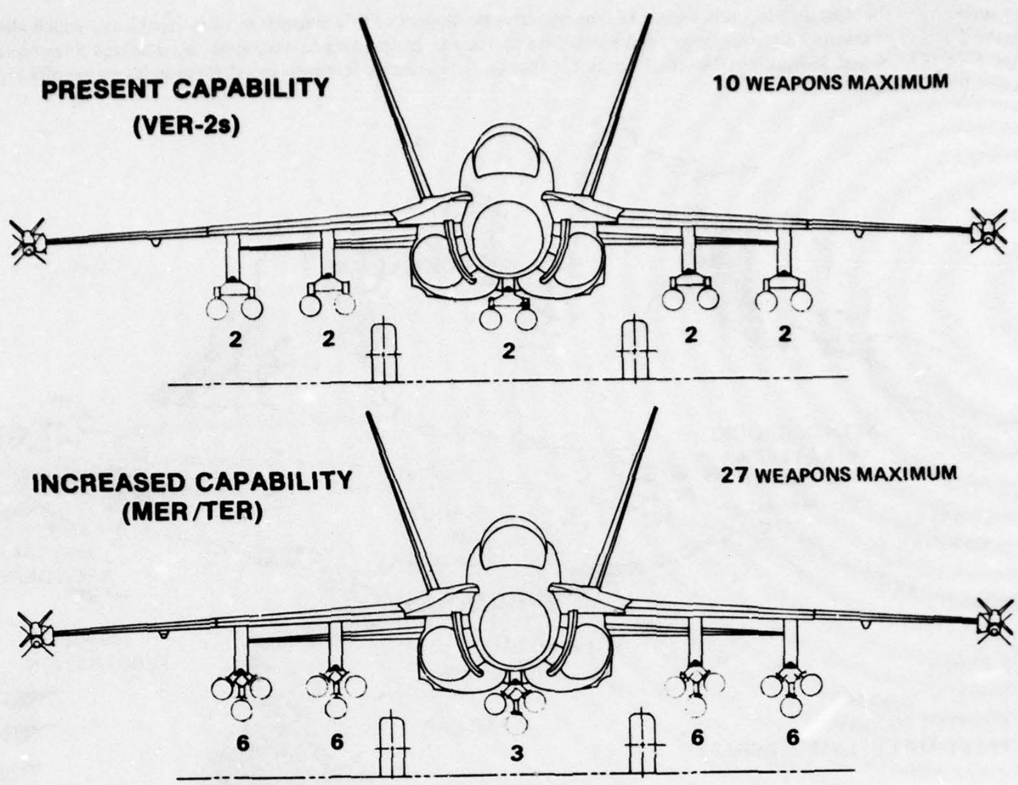

FIGURE 15
F-18 ADDITIONAL WEAPON CAPABILITY

Multiple Mission Adaptability

The F-18 Navy fighter and A-18 Navy light attack aircraft are almost indistinguishable (Figure 16). One cockpit display in the fighter version, the horizontal situation indicator, is replaced by a moving map display in the attack version. Otherwise, the two aircraft differ only in externally-carried mission equipment, and are readily reconfigurable. Medium-range missiles (MRM) carried in the fighter configuration are replaced by a forward-looking infrared sensor and a laser spot tracker in the light attack configuration.

NAVY FIGHTER F-18	2 UHF RADIOS	HORIZONTAL SITUATION INDICATOR	ALL OTHER COMPONENTS IDENTICAL	
NAVY ATTACK A-18	2 UHF RADIOS	MOVING MAP DISPLAY		
MARINE FIGHTER F-18	1 UHF 1 VHF	HORIZONTAL SITUATION INDICATOR		
TRAINER TF-18	2 UHF RADIOS	HORIZONTAL SITUATION INDICATOR	400 LB LESS INTERNAL FUEL	2 SEATS

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FIGURE 16
TECHNOLOGY PERMITS 99+% COMMONALITY
F-18 FIGHTER - A-18 ATTACK

A nominal carrier deck load will inclde one squadron of 12 F-18 fighters, and two squadrons totaling 24 A-18 light attack aircraft, replacing one squadron of F-4 fighters and two squadrons of A-7 light attack aircraft. The carrier complement will normally be rounded out by one squadron of F-14 fighters for fleet air defense, plus A-6 medium attack aircraft and utility aircraft. Reconfigurability of the F-18 and A-18 permits allocation of all 36 aircraft to the fighter role if combat exigencies so dictate, or alternative allocation of all 36 to the light attack role. Thus, in a rapidly changing tactical environment, 36 dual-role aircraft provide the same capability as up to 72 dedicated-mission aircraft.

We have analyzed a scenario (Figure 17) in which a carrier task force (two aircraft carriers) has the mission of power projection ashore. The primary objective is attack of ground targets. In the early stages of a 30-day campaign, enemy fighter opposition requires a strong fighter escort force for the light attack aircraft. If the F-18 squadrons are incapable of meeting the desired sortie rate, because of either attrition or maintenance delays, A-18's are reassigned as fighters. Once the enemy local air opposition has been depleted, the F-18's are

assigned to supplement the light attack force. Figure 18 demonstrates the impact of this multiple mission capability, which allows an increase in the fraction of scheduled sorties flown, and a one-third increase in ground targets destroyed, as compared to an equal force of dedicated F-18 fighters and dedicated A-18 attack aircraft. Other quite reasonable scenarios can show even larger payoffs for mission flexibility as a force multiplier.

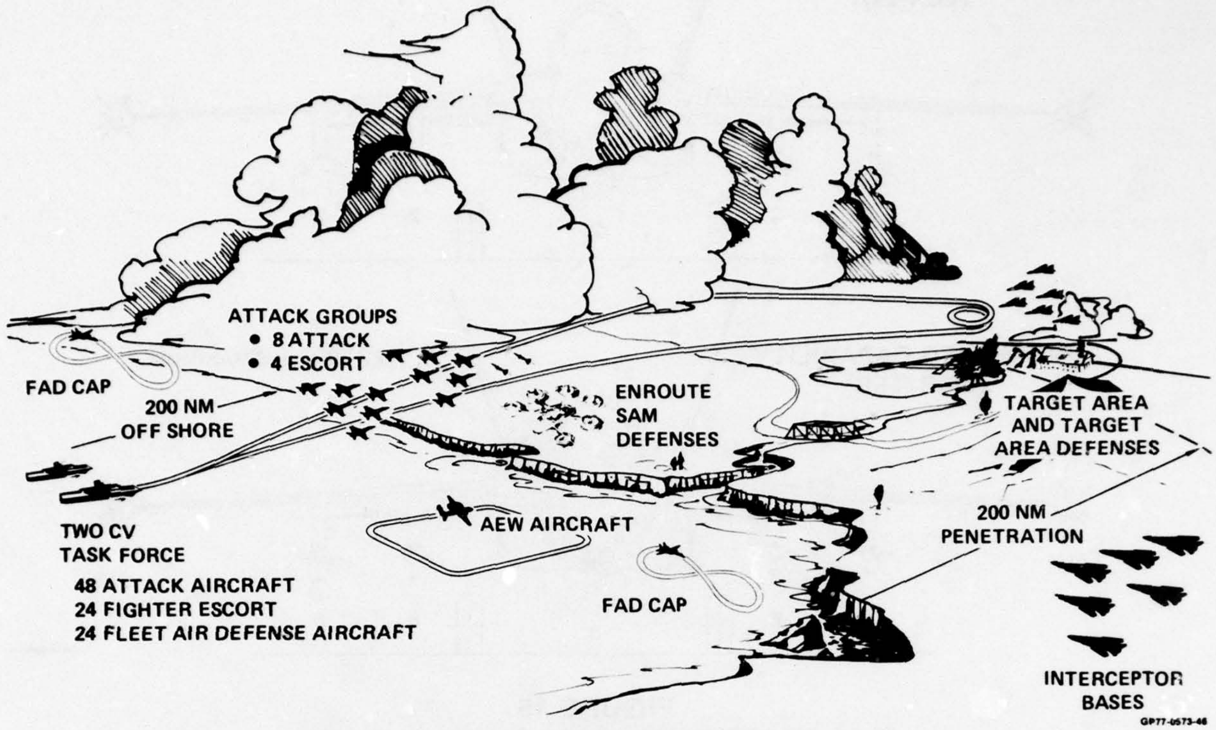

FIGURE 17
POWER PROJECTION SCENARIO

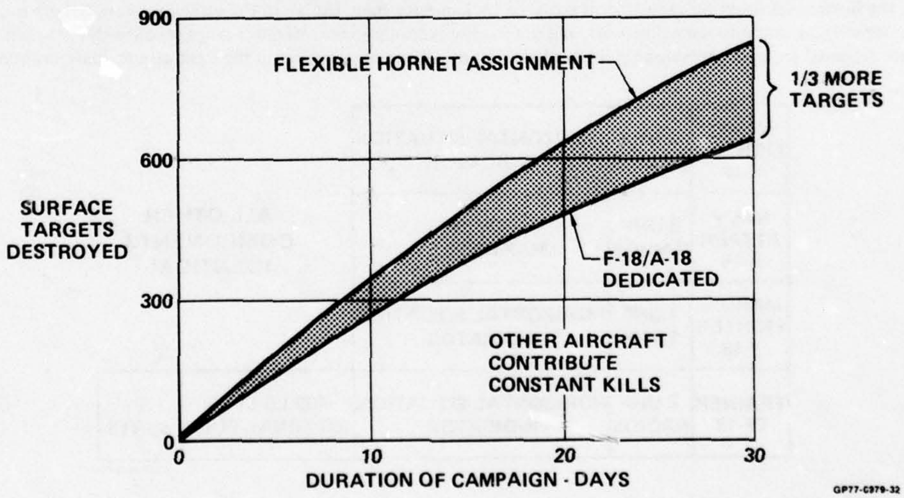

FIGURE 18
THE ADVANTAGES OF BEING FLEXIBLE

Availability

At times, during peacetime deployments, Navy fighter aircraft have been available - ready to fly with full mission capability - no more than 40 percent of the time. Space board an aircraft carrier limits the number of maintenance personnel and the supply of spare parts carried, placing an upper limit on the availability of an aircraft with fixed reliability and maintainability characteristics. Further improvements must come from design characteristics that enhance reliability and maintainability or reduce turnaround time.

Figure 19 shows Hornet reliability and maintainability projections, as compared to the current F-4 and A-7 levels. The cross-hatching shown for the Hornet shows the range of values expected to be realized in normal fleet operations; better values are required to be demonstrated, and the prime contractor, subcontractors, and suppliers have design requirements that exceed the projected fleet reliability by a factor of approximately two.

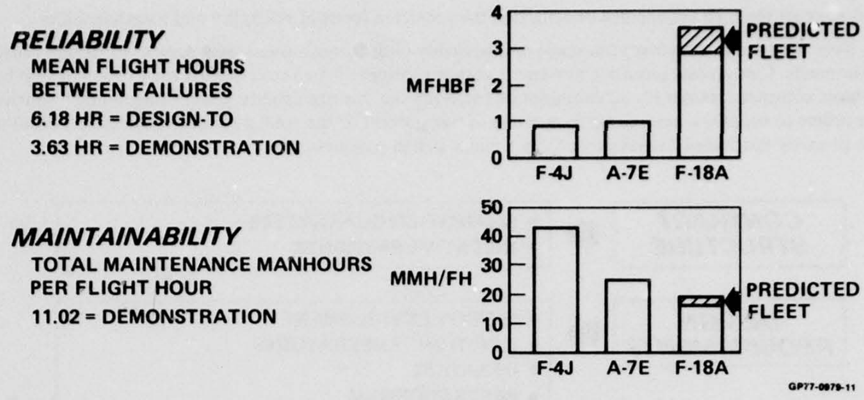

FIGURE 19
HORNET - A REALIZABLE R&M PROGRAM

Figure 20 translates those gains to increased availability. Availability is a direct force multiplier, its payoff being a function of the sortie-rate demand. Using the same scenario in which we assessed the payoff for multiple mission capability, we find (Figure 21) that the increase in availability over the F-4 and A-7 increases combat effectiveness by 31 percent, as measured by ground targets killed. More ground targets are destroyed as more attack sorties are flown. In addition, the higher sortie rate results in a more favorable force ratio against the enemy fighter threat, defeating the fighter threat more rapidly, with fewer friendly losses.

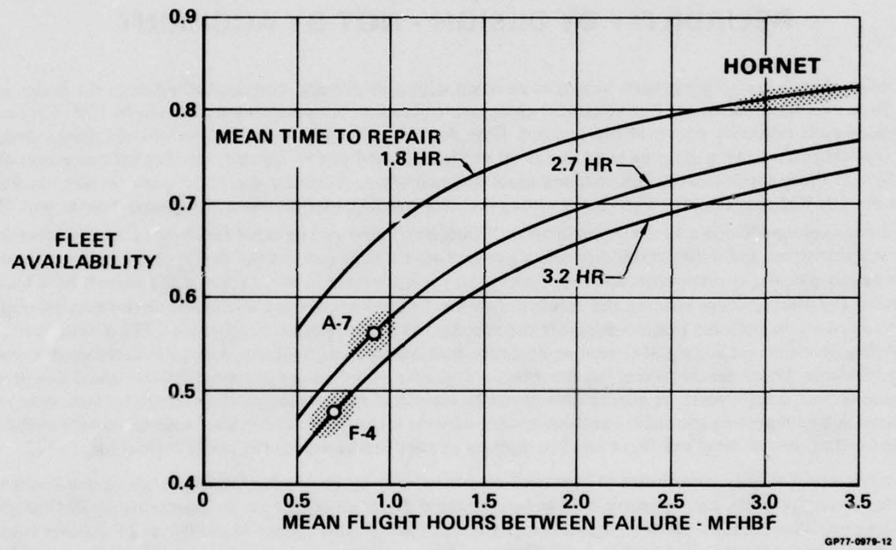

FIGURE 20
SHORT REPAIR TIME HELPS AVAILABILITY

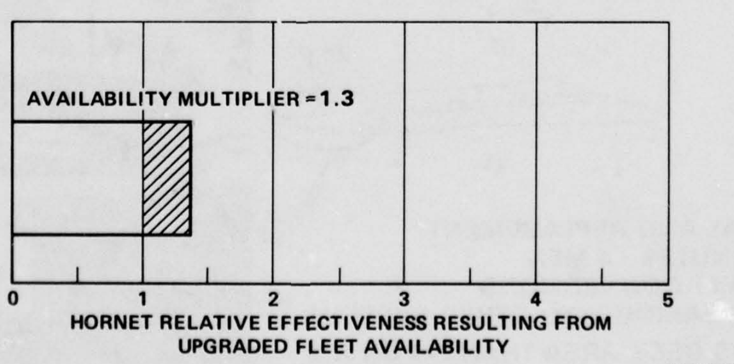

FIGURE 21
AVAILABILITY CONTRIBUTIONS TO POWER PROJECTION MISSION EFFECTIVENESS

Three basic factors support the high probability of achieving the projected levels of reliability and maintainability.

First, (Figure 22) Navy program management has made unmistakably clear its seriousness with respect to Hornet reliability and maintainability requirements. Contractual incentive provisions permit earnings of the prime contractor to be increased by as much as 30 percent over the basic contract earnings for achievement of reliability and maintainability goals. Comparable incentives have been included in purchase orders to suppliers, amounting to as much as five percent of the total purchase order values, to assure that all levels of industry are properly motivated to make reliability a major design parameter.

CONTRACT STRUCTURE ➡
- QUANTIFIED GUARANTEES
- INCENTIVE PAYMENTS

DESIGN REQUIREMENTS ➡
- MISSION ENVIRONMENT
- JUNCTION TEMPERATURES
- DERATING
- PARTS PROGRAM
- WEIGHT/CAPABILITY TRADED
- RELIABILITY ANALYSIS

REALISTIC TESTING ➡
- DEVELOPMENT THROUGH TEST-ANALYZE-FIX
- REALISTIC MISSION ENVIRONMENT
- LABORATORY AND FLIGHT DEMONSTRATION
- CLOSED LOOP FAILURE TRACKING

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FIGURE 22
RELIABILITY BY DESIGN - NOT BY ACCIDENT

Secondly, reliability demonstrability has been treated as an equal with performance, cost, and schedule in the design process. Only examples can be given in a brief discussion. For electronic equipments, junction temperatures are limited to 100° Celsius, providing significantly increased parts reliability over previous practice. Firm derating criteria have been applied to all avionics designs, with some relay and switch applications involving derating to as low as 10 percent of rated power. Overall, the thermal and power derating criteria compare favorably with those applied in the U.S. manned space program where reliability was paramount. Weight penalties have been incurred in the interest of higher reliability, when the resulting performance degradation was not disproportionate with the reliability gain.

Reliability predictions are made down to the component or "black box" level and updated for design changes. Electrical and thermal as well as structural stress analyses and failure mode and effect analyses are made to support the design process. The temperature, altitude, humidity, vibration, and acoustic environments and exposure cycles for equipment in every zone of the aircraft have been identified by analysis of U.S. Navy and Marine Corps training and combat missions. From this analysis a composite environmental design mission cycle was developed and imposed on both the prime contractor and suppliers as a specification requirement. The development program then provides for reliability development testing of critical equipments, that testing being performed in a simulated operational mission environment to the extent feasible. Development testing has the primary objective of inducing equipment failures under design environmental and operating conditions so that design, parts, or quality defects can be identified and eliminated prior to production. A key part of this program is a closed loop failure reporting and failure analysis system wherein standard procedures are used to provide visibility throughout the development qualification, system level and flight test activities by all suppliers as well as the prime contractor.

Increased reliability automatically contributes to improved maintainability by decreasing the frequency of maintenance actions. Design emphasis on access for maintenance, and extensive built-in-test for rapid diagnosis contribute to decreasing repair time given a failure. Figure 23 illustrates provisions for rapid Hornet engine change, for example. Engine change in as little as 21 minutes is to be demonstrated; normal fleet operations cannot be expected to match the demonstration capability, but are expected to provide a manifold reduction from the F-4's engine-change time. Figure 24 summarizes Hornet access provisions for improved maintainability.

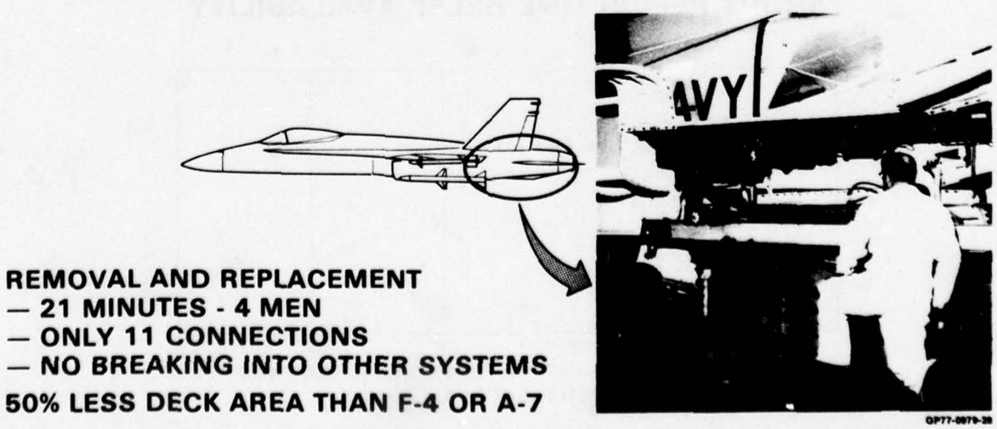

FIGURE 23
EASY REMOVAL AND REPLACEMENT
ENGINE

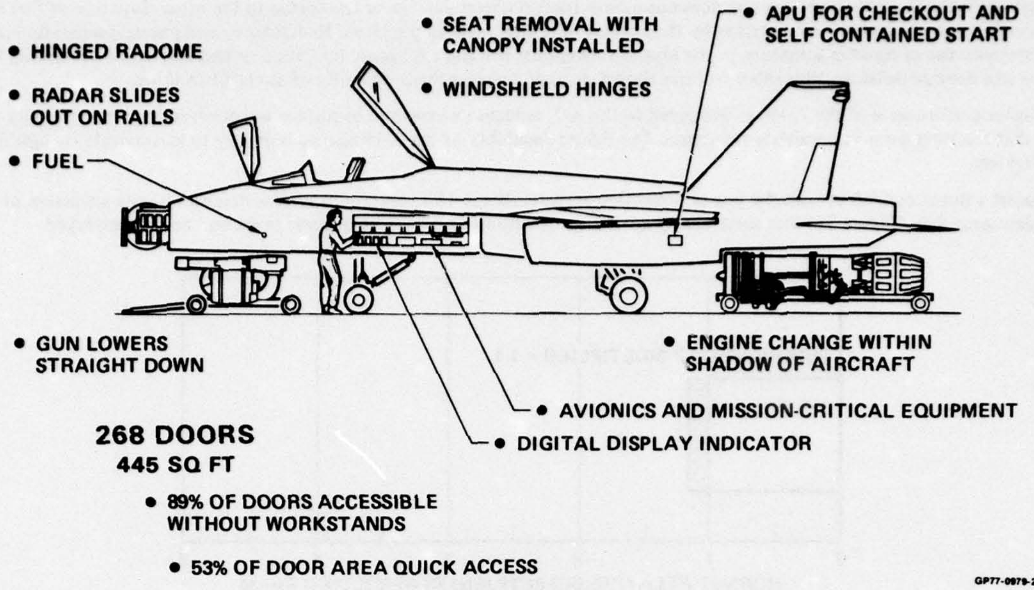

FIGURE 24
THE F-18 IS DESIGNED FOR EASY ACCESS

Survivability

Increased survivability against threat weapons is a force multiplier, increasing the combat life of the aircraft and permitting it to generate more combat sorties. Many U.S. combat aircraft were modified, as a result of early experience in Southeast Asia, to decrease the vulnerability of flight control systems and fuel, in particular.

Obviously, designers can go much further in hardening new fighter designs against conventional weapons than is efficient in modifying an existing design. Figure 25 illustrates some of the vulnerability-reduction features designed into the F-18 Hornet. Design for survivability begins with the basic aircraft configuration. Twin-engine safety becomes a contributor to increased combat survivability when the engines

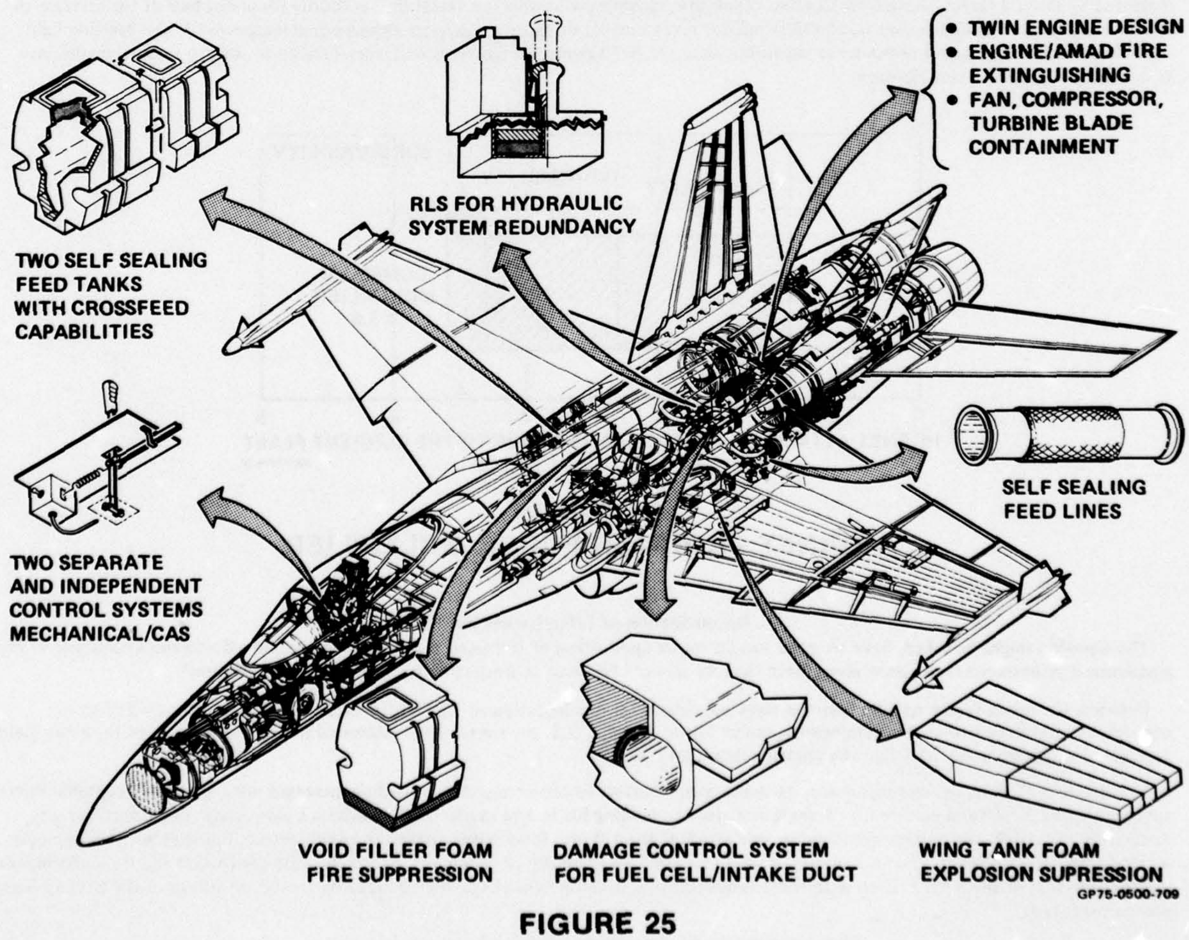

FIGURE 25
SURVIVABILITY THROUGH DESIGN

are effectively isolated, so that a hit on one does not permit foreign object damage or fire spread to the other. Isolation of fuel from the engine section protects against fuel ingestion by the engines following fuel-cell puncture. Redundancy and physical separation of flight control systems, use of rip-stop actuators, protection of fuel against fire and explosion, hydraulic system reservoir level sensing for leak detection and damage isolation, plus other features shown, further decrease the probability of aircraft loss if hit.

The higher performance of the A-18, as compared to the A-7, reduces its exposure to surface weapons, adding a survivability increment beyond that resulting from vulnerability reduction. The fighter capability of the A-18 also adds greatly to its survivability against enemy fighters.

Against a threat consistent with the power projection scenario, the A-18's loss rate to surface defenses is only a fraction of the A-7's. In a 30-day campaign, (Figure 26) that survivability advantage contributes a 9 percent increase in ground targets destroyed.

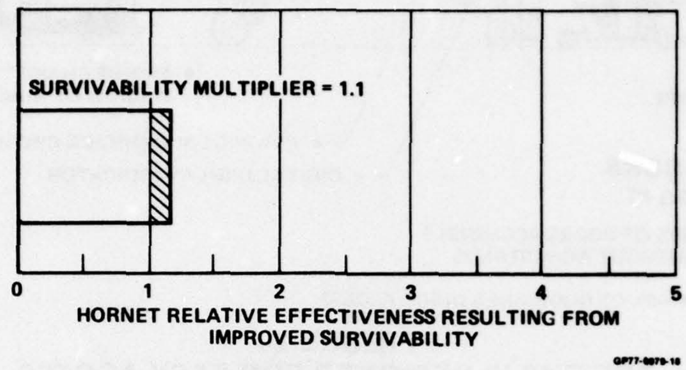

FIGURE 26
SURVIVABILITY CONTRIBUTIONS TO POWER PROJECTION MISSION EFFECTIVENESS

Combined Effect of Force Multipliers

When combat performance and lethality, multimission versatility, availability, and survivability benefits are combined in the force projection scenario (Figure 27), the Hornet carrier complement proves to be 3.6 times as effective as the F-4J/A-7 complement, as measured by ground targets destroyed. Combat capability, including multimission versatility, accounts for about half of the increase in effectiveness, with availability and survivability against surface-to-air threats providing an almost equal increment. If the baseline light attack aircraft had had better self-defense *capability than the* A-7 against the fighter threat, survivability to surface threats would have made a more significant contribution.

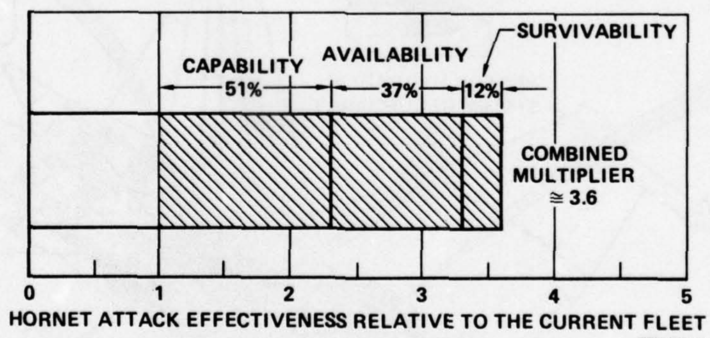

FIGURE 27
HORNET - A MAJOR FORCE MULTIPLIER

Reconciliation of Effectiveness and Costs

The question might be asked, have we gone too far in the application of advanced technology to combat effectiveness multipliers? Have peacetime requirements really been given more than lip service? Has cost of ownership been a major consideration?

Cost was the major factor which restricted Navy consideration to adaptations of the YF-16 or YF-17. Those lightweight fighter prototypes were design-to cost developments, under contract to the U.S. Air Force. Each addressed minimum-cost design to satisfy fighter performance requirements to defeat the maneuvering fighter threat.

Modification (Figure 28) was mandatory to make either prototype carrier-suitable, including increased wing area for acceptable carrier approach speed, structural beefup for catapult and arrested landing loads, and carrier-landing avionics aids. Range extension also was mandatory, to satisfy carrier requirements for safe standoff from shore. Reasonable growth in engine thrust, together with the increase in wing area, only partially offset the weight increase required by the Navy environment, rather than driving toward fighter performance greater than that of the YF-17. Even with those modifications, development costs were reduced by taking advantage of the already-flight-proven prototype.

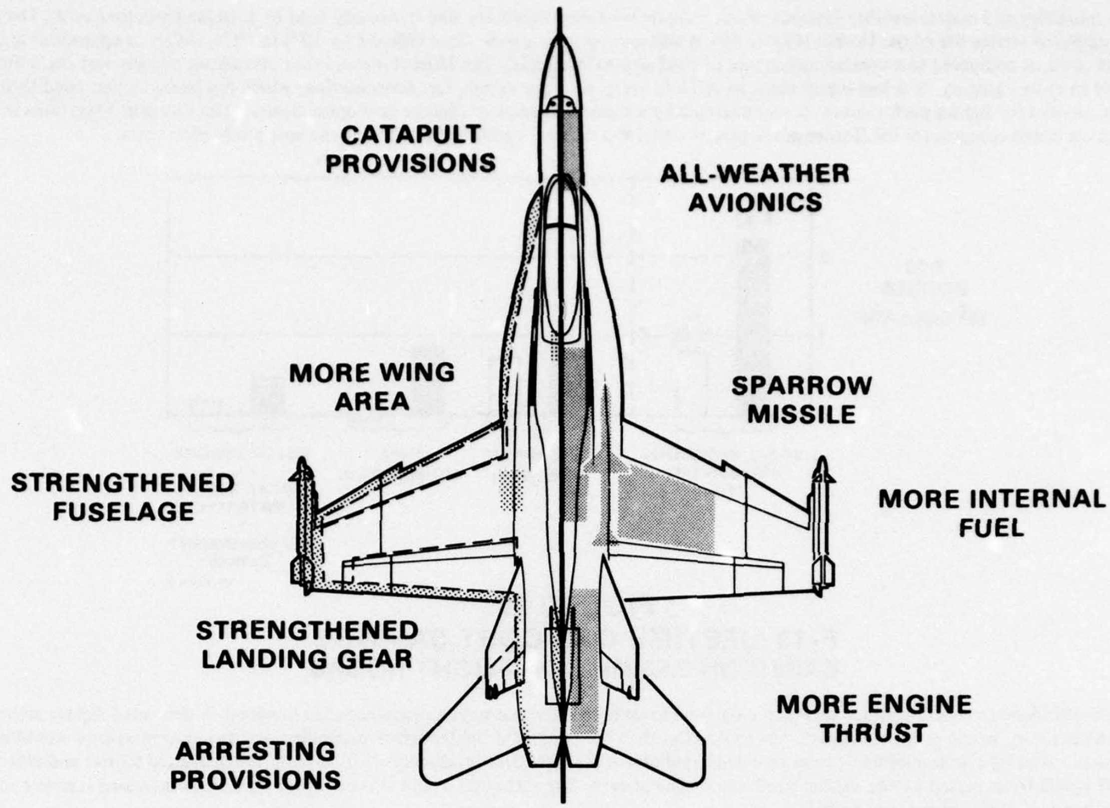

FIGURE 28
YF-17 + CARRIER SUITABILITY +NAVY MISSION = F-18

As regards fighter capability, the major difference between the YF-17 and the F-18 is beyond-visual-range AIM-7F missile capability of the Hornet, and a fire control radar permitting full-envelope employment of that missile against threat aircraft at all altitudes. Before full scale development was authorized, trade studies were conducted (Figure 29), examining closely the costs of AIM-7F capability and of radar range increments, together with their force-multiplier effectiveness. The beyond-visual-range missile provides kill opportunities before closing to maneuvering combat, thus improving the exchange ratio against even dayfighter threats. Radar range adequate to permit maximum range missile launches becomes more significant as the fraction of the threat equipped with beyond-visual-range missiles increases. The force multiplier benefits fully justified the modest cost increases.

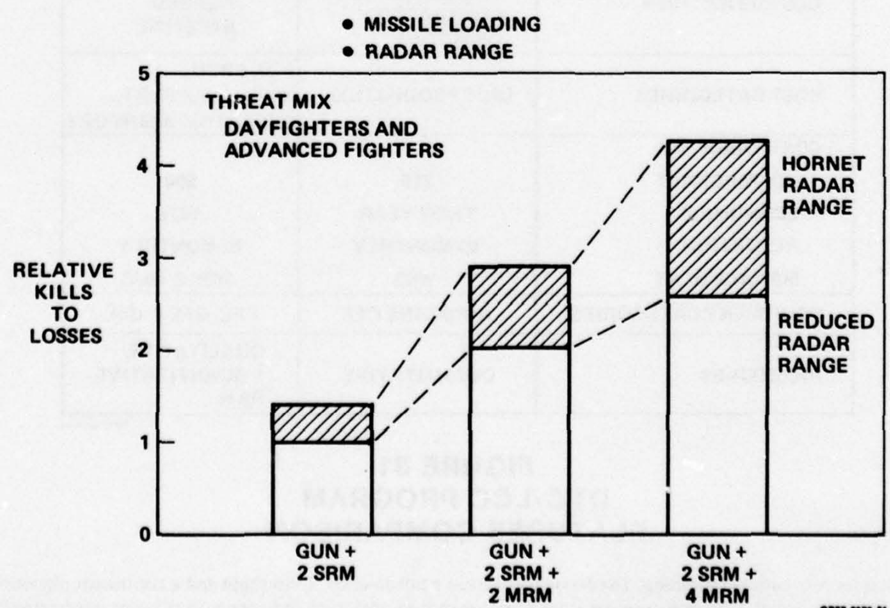

FIGURE 29
EFFECTIVENESS SENSITIVITY

The reliability and maintainability features which increase wartime availability simultaneously tend to decrease peacetime costs. Over the anticipated service life of the Hornet (Figure 30), it will save approximately three billion (3×10^9) in 1976 dollars in operations and support costs, as compared to a corresponding mix of F-4J and A-7E aircraft. The Hornet shows lower operations and support costs than the F-4J in every category. It is less costly than the A-7E in every category except fuel consumption, where the Hornet's improved thrust loading, needed for fighter performance, is accompanied by a modest increase in lifetime fuel consumption. The saving in operations and support costs will compensate for Hornet development costs, and offset a significant portion of the unit production costs.

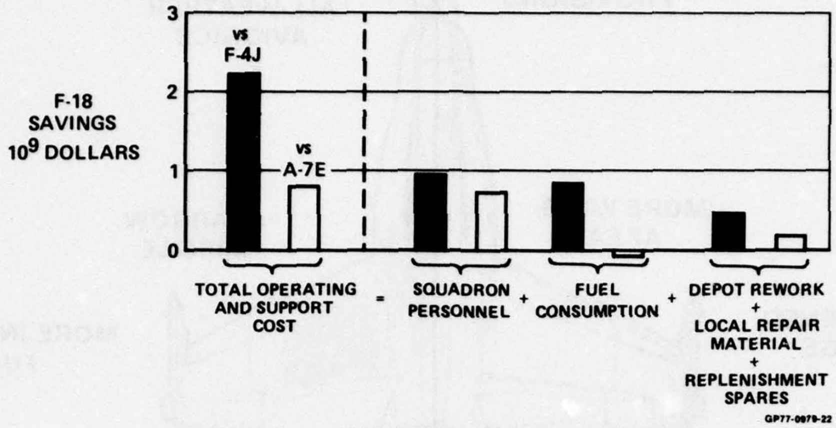

FIGURE 30
F-18 LIFETIME O&S COST SAVINGS
BASED ON 2.63 MILLION FLIGHT HOURS

The multimission versatility which provides a wartime force multiplier also pays a peacetime cost dividend. A dedicated fighter aircraft of F-18 capability would cost only slightly less to develop than the combined fighter-attack capability, and the modest savings would be more than offset by the development costs of a dedicated attack aircraft. Unit production costs of both the dedicated fighter and attack aircraft would be increased by the smaller production runs of each. Support costs would also be higher because of increased logistics costs to maintain two different types of aircraft.

The Hornet basic design concepts thus include multimission, reliability, maintainability, and combined safety/survivability features which contribute both to the need for wartime force multipliers and the necessity for reduced peacetime costs. Increased combat capability, and survivability features that provide no collateral peacetime safety benefits, have been required to justify their cost impacts by demonstrating force multiplier effects that far outweigh their cost increments. This concept properly balances peacetime and wartime requirements. The remaining requirement is to manage detailed design and procurement decisions with equally painstaking consideration of all factors that influence peacetime costs and combat effectiveness. That management program is known as design-to-cost/life cycle cost, and is outlined in Figure 31.

	DESIGN TO COST	LIFE CYCLE COST
COST OBJECTIVES	NAVY - DEFINED THRESHOLD AND CONTRACTOR OBJECTIVE	MUTUALLY AGREED BASELINE
COST CATEGORIES	UNIT PRODUCTION	FSD, PROD, INITIAL SUPPORT, OPERATING & SUPPORT
COST TRACKING		
NO. AIRCRAFT	219	800
ECONOMICS	THEN YEAR	1975
REPORTING	BI-MONTHLY	BI-MONTHLY
MANAGEMENT	WBS	WBS & WUC
EQUIPMENT CATEGORIES	AIRBORNE CFE	CFE, GFE & GSE
INCENTIVES	QUANTITATIVE	QUALITATIVE + QUANTITATIVE $\overline{R}\&\overline{M}$

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FIGURE 31
DTC/LCC PROGRAM FEATURES COMPARISON

Design-to-cost has its conventional meaning. The Navy has defined a not-to-exceed threshold and a contractor objective for unit production costs of the first 219 aircraft. Cost management is by work breakdown structure and covers all airborne contractor-furnished equipment. Incentive awards and penalties are contractually specified. Life cycle cost tracking and trade studies encompass the costs of full scale development, production, initial support, and operations and support costs, applicable to the total planned procurement of 800 aircraft, and including government-furnished equipment and ground support equipment in addition to airborne contractor-furnished

equipment. An appreciable incentive award fee can be earned through life cycle cost management. That incentive fee is in addition to the incentives for quantitative reliability and maintainability achievement, which contribute directly to operations and support cost reductions.

Trade studies completed through early 1977 (Figure 32) resulted in decisions with cost avoidance payoffs approximating $220 million. Production cost avoidance accounted for approximately two-thirds of that total.

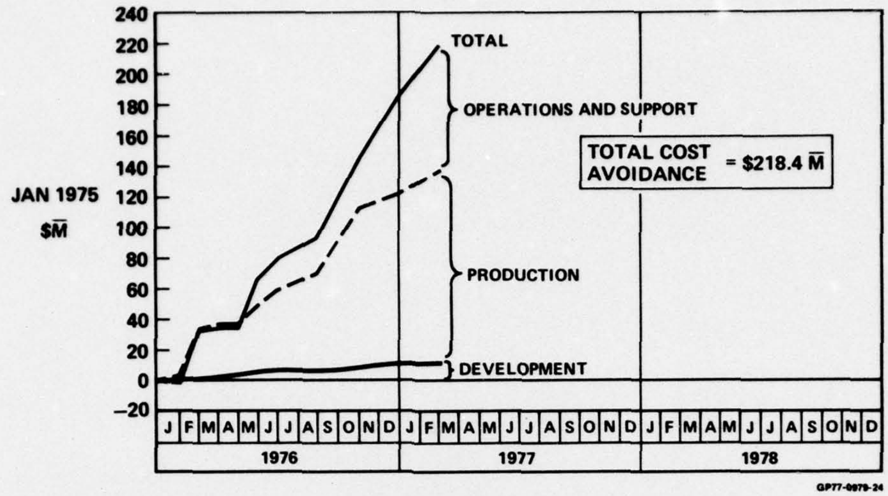

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FIGURE 32
HORNET LIFE CYCLE COST AVOIDANCE
RESULTING FROM TRADE STUDIES

The impact of life cycle cost tracking on procurement decisions is illustrated in Figure 33, based on negotiated procurements totaling almost half of procured equipment cost. The figure uses as a baseline the costs that would have been incurred if vendor selection had considered only the traditional factors of performance and weight. The bars show the cost savings that would have resulted if vendor selection had emphasized production costs (the design-to-cost emphasis), reliability, maintainability, or life cycle cost. Actual vendor selections were most consistent with the life cycle cost emphasis, with total savings approximating 90 percent of the potential savings.

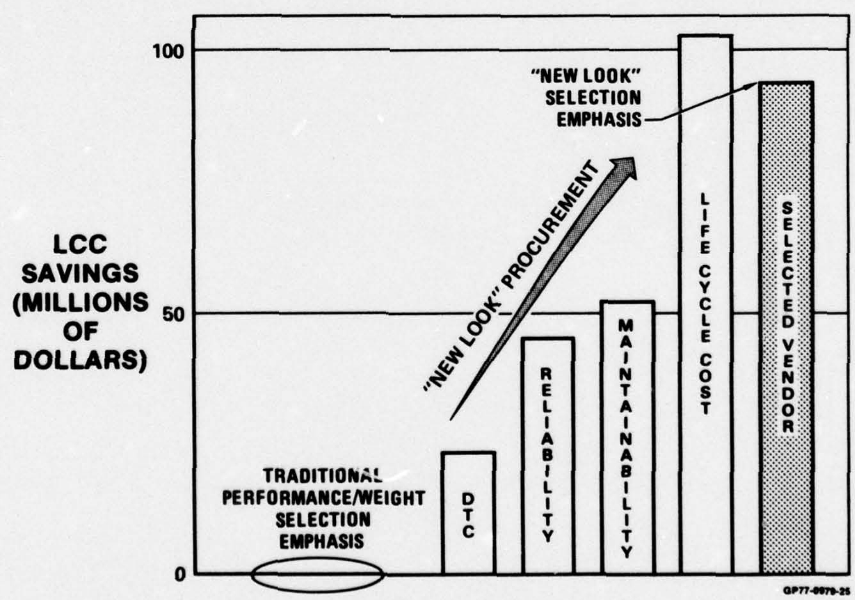

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FIGURE 33
HORNET PROCUREMENT RECORD - LCC SUCCESS
BASED ON 46 PERCENT OF PROCURED CFE COST

This record suggests, accurately, that procurement decisions for the Hornet have weighed all elements of life cycle cost, and have favored, on balance, the selections that minimize life cycle cost. At the same time, the selection process has not considered life cycle costs as the sole selection criterion, but has chosen some more costly alternatives where gains in performance, availability, or weapon system capability showed sufficiently strong justification.

Where do we go from here? We see no indication that potential adversaries are inclined to reduce their force sizes, and every indication that they will continue to improve the quality of those forces. Significant increases in the real purchasing power of NATO defense budgets are not likely; increases of three percent annually are being urged, implying a bound on affordability.

We shall continue to need force multipliers and some of those multipliers lie in the realm of tactics, training, logistics, and command control systems. But the fighter design team must remain conscious of the necessity for designed-in force multipliers of the kinds discussed herein: superior combat performance, lethal firepower, multimission versatility, availability, and survivability. We have probably not yet struck the optimum balance of those force multipliers against one another, or of their combination against costs. Technological advance, while increasing our design options, will make the tradeoffs even more difficult. But performing those tradeoffs as intelligently and systematically as possible will remain the key to providing superior and affordable fighters by design.

AERODYNAMIQUE DE LA NOUVELLE GENERATION D'AVIONS DE COMBAT A AILE DELTA

par

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1. Cette conférence n'est pas une présentation détaillée du MIRAGE 2000 mais une réflexion sur l'utilisation de la formule aérodynamique de l'aile en delta pour les avions de combat modernes.

2. Le premier vol d'un avion MARCEL DASSAULT équipé d'une aile Delta remonte à 1955. Il s'agissait du petit MIRAGE 1 propulsé par deux réacteurs VIPER. Il était le premier de la lignée des 1500 MIRAGE actuels.

Les premiers MIRAGE 3 sont sortis en série en 1961 propulsés par un réacteur ATAR 9. Après 17 ans, cet avion continue de recevoir des commandes, contrairement à tous les autres avions de sa génération (Tiger, F104, etc...).

Sa formule aérodynamique, aile en Delta de 60° de flèche au bord d'attaque, a permis le vol à Mach 2 avec un seul réacteur de 6 tonnes de poussée alors que ses concurrents étaient équipés d'un J79 de 7 tonnes.

La contrepartie de cette performance supersonique a été une augmentation importante de la vitesse d'approche, 180 kt au lieu des 140 kt pour la famille des MYSTERE. Cette contrainte a été acceptée pour les MIRAGE III mais les fiches programmes ultérieures de l'Etat Major de l'Armée de l'Air Française ont ensuite demandé une vitesse d'approche inférieure à 150 kt.

Ceci explique la naissance du MIRAGE F1. Le développement de l'ATAR à 7,2 tonnes (ATAR 9K) permettait en effet d'admettre, sans dégradation des performances supersoniques, l'augmentation de la trainée de l'aile en flèche avec empennage arrière. Par contre, l'aile avec empennage pouvait être hypersustentée et on pouvait revenir aux 150 kt. Cet avion est sorti en série en 1971 et bénéficie de nombreuses commandes.

La planche 1 qui ne fait figurer comme paramètre que la vitesse d'approche indique néanmoins, sur une période de 30 ans, le placement relatif des principaux avions de combat sortis par les AVIONS MARCEL DASSAULT.

On note les vols pendant quelques années des avions MIRAGE G et G8 à aile à géométrie variable dont la vitesse d'approche était dans la gamme des 125 kt, ce qui permettait l'appontage sur les porte-avions français. Les avions à géométrie variable ont été arrêtés en 1973 pour des raisons de coût et surtout pour leur infériorité dans les missions de supériorité aérienne.

Le MIRAGE G8 fut remplacé par le MIRAGE G8A biréacteur SNECMA M53 à géométrie fixe dont le prototype fut annulé fin 1975 pour des raisons budgétaires.

L'année 1975 est une date clef pour le retour à l'aile Delta. Elle est figurée par une grosse flèche noire sur la planche 1.

Pour des raisons qui seront expliquées dans la suite de cette conférence, un avion à aile Delta était alors capable d'avoir une vitesse d'approche de 150 kt. Comme, d'autre part, les contraintes budgétaires en France amenaient à ne pouvoir construire qu'un monoréacteur M53 de 9 tonnes de poussée au lieu d'un biréacteur disposant de 18 tonnes de poussée, l'aile Delta était la formule aérodynamique qui permettait de minimiser la dégradation des performances supersoniques demandées au MIRAGE G8A.

Après la décision gouvernementale de remplacer le biréacteur MIRAGE G8A par le monoréacteur MIRAGE 2000, plusieurs prototypes sont en cours de construction et la sortie des MIRAGE 2000 de série est prévue en 1982.

3. Dans la décision prise en France fin 1975, nous devons mentionner la concurrence de plusieurs facteurs. Ils figurent sur la planche 2.

Un contrat du Service Technique Aéronautique comparant des formules d'avions CCV a montré l'intérêt très important des centrages arrières sur une aile Delta.

Les premiers vols du réacteur M53 sur le prototype MIRAGE F1 E ont montré, grâce à un meilleur temps de réponse par rapport à l'ATAR, que les incidences d'approche pouvaient être augmentées.

Finalement, les AMD-BA avaient étudié en soufflerie sur leurs fonds propres de nouvelles ailes en Delta.

4. Pour passer du MIRAGE 3 à 180 kt au MIRAGE 2000 à 150 kt (malgré un système d'armes plus lourd), il fallait utiliser plusieurs facteurs car aucun facteur, à lui seul, n'était suffisant pour apporter ce gain.

Nous en mentionnons 3 sur la planche 3 :

1/ Le centrage arrière, permis par la technologie des commandes de vol électriques.

2/ L'incidence d'approche, augmentée grâce au meilleur temps de réponse du réacteur M53.

3/ La charge alaire diminuée, donc une surface plus grande.

Ce point sera vu plus en détail dans la suite.

5. Nous venons de voir que nous avons besoin d'augmenter la surface de voilure. Ceci devrait être facile car nous disposons d'un réacteur de 9 tonnes, au lieu de 6, pour le MIRAGE 3.

Néanmoins les demandes opérationnelles pour la mission de défense opérationnelle sont très dures. N'oublions pas que cet avion remplace un intercepteur lourd qui disposait de 18 tonnes de poussée!!

Les principales demandes opérationnelles ayant une répercussion sur la trainée en supersonique figurent sur les 4 premiers points de la planche 4.

6. Cependant l'ensemble des demandes opérationnelles supersoniques était tel que la surface d'aile ne pouvait pas être suffisamment augmentée pour atteindre les 150 kt.

Mais nous avions étudié en soufflerie plusieurs cambrures de bord d'attaque (planche 5). Il faut que nous rappelions que le MIRAGE 3 n'est pas équipé de dispositif de bord d'attaque mobile, par contre le profil du MIRAGE 3 a une forte cambrure de bord d'attaque, surtout en bout d'aile, réalisant un compromis entre les missions d'interception et celles de supériorité aérienne.

C'est pourquoi nous avons décidé que le profil du MIRAGE 2000 bénéficierait d'une légère cambrure limitée à une valeur telle que le supplément de trainée en supersonique ne soit pas important. Par contre, ce choix imposait l'installation des becs mobiles de bord d'attaque.

7. Ces becs permettent en combat non seulement d'égaler les performances de la voilure cambrée du MIRAGE 3 mais encore de les augmenter considérablement, ainsi que le montrent les polaires de la planche 6.

On notera que les becs du MIRAGE 2000 sont toutes positions et fonction de l'incidence et du Mach afin de placer l'avion en position de trainée minimum (courbe enveloppe en pointillé sur la planche 6).

8. Par ailleurs, les méthodes théoriques de calcul aérodynamique tridimensionnel disponibles aux AVIONS MARCEL DASSAULT-BREGUET AVIATION ont permis d'optimiser l'intégration des formes de voilure et de fuselage.

La planche 7 montre une coupe très typique du MIRAGE 3 et du MIRAGE 2000. Les nouvelles formes d'emplanture ont permis, par leur hauteur d'attaches de longeron augmentée, un gain de masse de voilure et ce, sans perte visible de trainée en supersonique.

En ajoutant à ce gain de masse celui provenant des élevons réalisés en fibre de carbone, l'aile du MIRAGE 2000 est plus légère que celle du MIRAGE 3 malgré une surface plus grande d'environ 20 % et l'adjonction de becs mobiles (voir planche 8).

9. Nous allons maintenant quitter les domaines des vitesses d'approche et des vitesses supersoniques pour nous intéresser aux domaines des grandes incidences en transsonique (limites de manoeuvre).

Nous allons utiliser à cet effet les courbes de stabilité longitudinale (C_m, C_z) de plusieurs avions (conventions de signe francaises).

Nous avons porté sur la figure de gauche de la planche 9, en trait plein, une courbe de stabilité en transsonique typique du MIRAGE F1 de série. On note, à partir d'un certain C_z, une augmentation importante de stabilité (hyperstabilité) typique des avions à aile en flèche à empennage bas.

Dans une version CCV du MIRAGE F1, la marge statique aurait été négative à faible C_z mais on aurait retrouvé comme précédemment, à partir d'un certain C_z, une hyperstabilité c'est-à-dire, à forte incidence, une perte de C_z équilibré car l'empennage doit équilibrer ce fort couple piqueur.

La présence d'Apex à la racine de la voilure permet de remonter le C_z de cassure de stabilité (voir figure droite de la planche 9), c'est pourquoi on a vu des Apex sur le MIRAGE G8 A ou sur le F16. Mais le défaut est repoussé mais pas supprimé.

Dans le cas d'un MIRAGE 3, on avait une courbe de stabilité différente (voir courbe en trait plein de la figure de gauche de la planche 10).

Une forte hyperstabilité obtenue pratiquement à C_z constant était suivie par, à fort C_z, une légère instabilité. Toujours sur la figure de gauche, nous avons porté le MIRAGE avec becs sortis mais sans aigrettes placées au-dessus et en avant de la voilure. On note la stabilité négative d'un avion CCV et un gain important de C_z d'hyperstabilité par rapport au MIRAGE 3. Mais l'étude en soufflerie de différents dispositifs a abouti à la définition d'une aigrette permettant (figure de droite de la planche 10) par rapport au MIRAGE 3 :

- une augmentation de 70 % du C_z de limite de manoeuvre
- une réduction très importante de l'hyperstabilité.

Nous ne quitterons pas cette planche 10 sans dire que le rôle aérodynamique de l'aigrette est très différent de celui d'un canard classique. Un canard aurait, en particulier, changé la marge statique comme l'indique la petite courbe en pointillé du bas de la figure de droite.

10. Si les études menées en longitudinal ont permis, *comme nous venons* de le voir, d'augmenter considérablement le domaine de C_z et d'incidence du MIRAGE 3, il fallait s'assurer que nous n'allions pas rencontrer à ces fortes incidences d'autres problèmes.

Nous avons ainsi vérifié, après études de différentes dérives , que l'incidence de perte de stabilité latérale était suffisamment élevée. La planche n°11, bien que trop simple pour représenter les qualités de vol latérales d'un avion à commande de vol électrique, montre cependant un gain en transsonique de 5 à 7° par rapport au MIRAGE 3.

Pour la même raison, nous avons travaillé les entrées d'air du MIRAGE 2000 pour assurer, à forte incidence, un écoulement satisfaisant le réacteur. Une entrée d'air ventrale aurait facilité, à forte incidence, le travail de l'aérodynamicien mais l'emport de charges ventrales importantes et sophistiquées aurait été limité d'une manière inacceptable. Les entrées d'air du MIRAGE, qui ont fait la preuve de leur bon fonctionnement dans les mains de nombreux utilisateurs, ont donc été conservées avec adjonction des dispositifs adaptés au vol à grande incidence. Les résultats d'essais en soufflerie figurent sur la planche 12 et font apparaître un gain de 5 à 7° d'incidence, homogène au gain précédent.

11. Si le principe des entrées d'air du MIRAGE a été conservé, par contre les performances en supersonique de ces entrées d'air ont été améliorées en travaillant grâce à des méthodes de calcul théorique sur ordinateur les formes du fuselage avant.

C'est ainsi que nous avons vérifié que, pour le même Mach de vol égal à Mach 2, le Mach local au niveau des entrées d'air passe de 2.10 pour le MIRAGE 3, à Mach 1.95 pour le MIRAGE 2000, ce qui se traduit bien sûr à Mach donné par un gain de rendement (voir planche 13).

12. Nous avons beaucoup parlé du rôle de l'aérodynamique théorique dans le développement de la formule aérodynamique du MIRAGE 2000. Mais il ne faut pas oublier que toutes les souffleries françaises ont été mises à contribution pour l'identification ou, dans certains domaines, la mise au point de cet avion.

A l'occasion de cet avion, de nombreuses modifications ont dû être apportées aux souffleries existantes et de nombreux dispositifs ont été réalisés. La principale raison a été l'exploration systématique des domaines de fortes incidences.

La liste des principales nouveautés mises en place dans les souffleries françaises figure sur la planche 14.

13. Nous avons évoqué les principales raisons du choix du MIRAGE 2000 et montré les principaux résultats acquis. Nous allons préciser maintenant ce choix en comparant les performances du MIRAGE 2000 à celles que nous aurions obtenues si nous avions retenu d'autres formules aérodynamiques et, en particulier, l'aile à géométrie variable et *l'aile en flèche avec empennage* arrière.

Notre premier critère de comparaison va être la limite de manoeuvre.

14. Tout d'abord pour nous recaler, nous allons comparer le MIRAGE 2000 à son grand frère, le MIRAGE 3 qui a déjà la réputation d'un excellent appareil de combat.

Nous utilisons, pour notre comparaison et les suivantes, un graphique montrant :

- le C_z maximum de combat
- la surface de voilure.

En effet, si tous les avions de notre comparaison ont une masse voisine (de l'ordre de 9 tonnes), leur limite de manoeuvre sera d'autant plus grande que le produit C_z max x Surface sera lui-même plus élevé ; elle sera donc proportionnelle dans chaque cas à la surface du rectangle hachurée (voir planche 15).

Dans le cas du MIRAGE 2000 comparé au MIRAGE 3 (planche 15), la surface étant augmentée de 20 % environ et le C_z max de 70 % comme nous l'avons vu sur la planche 10, la limite de manoeuvre est doublée.

15. Compte tenu de nos propres résultats obtenus en soufflerie et en vol bien sûr, relatifs aux avions à aile en flèche équipés de becs et de volets de combat et avec empennage arrière, et compte tenu également des performances d'avions d'autres avionneurs, on peut dire que, dans l'état actuel de l'art, un avion à empennage arrière bénéficie de 40 % environ de C_z max de plus.

Ceci s'applique aux avions à géométrie variable qui, en transsonique élevé et à fort facteur de charge, ne peuvent pas combattre avec l'aile en position avant mais amènent leur aile dans une position qui est dans la gamme des flèches des avions à géométrie fixe.

S'il y a un gain sur le C_z max, il y a par contre une perte importante sur la surface de voilure.Tous les avions à géométrie variable qui ont volé en France, aux Etats-Unis, en Angleterre/ Allemagne, ont une importante charge alaire.

Comme cette charge alaire est approximativement triple de celle du MIRAGE 2000, nous disons que si vous-même - ou un autre - dessinait un avion à géométrie variable autour d'un réacteur de 9 tonnes (et non de 11 tonnes comme pour le MIRAGE G-01), sa surface serait égale à 35 % de celle du MIRAGE 2000.

En effectuant le produit du rapport des C_z max (1,4) par le rapport des surfaces (0,35), on obtient qu'un avion de combat à géométrie variable avec même moteur aurait une limite de manoeuvre égale à 50 % à celle d'un MIRAGE 2000.

Ceci ramène un avion à géométrie variable au niveau du MIRAGE 3, ce qui n'est pas si mal mais insuffisant pour combattre contre les avions de la nouvelle génération.

Nous avons ainsi illustré la principale raison de l'arrêt des avions à géométrie variable en France.

16. De la même façon, nous avons imaginé un avion à aile fixe avec empennage arrière dessiné autour du même moteur de 9 tonnes. Nous l'avons baptisé sur la planche 17 le SUPER F1. Disons que son dessin et ses caractéristiques sont très voisins d'un avion américain récemment choisi par plusieurs pays européens.

On retrouvera le gain de 40 % de C_z max indiqué au paragraphe précédent. Mais là-aussi la présence d'empennage réduit la surface de voilure dessinable autour d'un fuselage donné.

Cet avion SUPER F1 aurait une surface de voilure inférieure de 35 % environ à celle du MIRAGE 2000.

On obtiendrait une limite de manoeuvre du SUPER F1 qui serait égale à 1,4 x 0,65 = 0,91 fois celle du MIRAGE 2000, soit légèrement inférieure.

Mais compte tenu que quelques pour-cent d'écart ne sont pas significatifs en raison des raffinements possibles de toute formule aérodynamique, examinons les performances supersoniques pour départager les 2 formules.

17. La planche 18 montre pour ces deux avions MIRAGE 2000 et SUPER F1, d'une part le produit S.C_z max qui est significatif du combat et, d'autre part le produit S.C_x significatif de la trainée totale de l'avion en supersonique.

On constate que, si les performances en combat sont voisines, comme nous venons de le voir, la trainée en supersonique du SUPER F1 est de 35 % supérieure à celle du MIRAGE 2000, à égalité de poussée de moteur, rappelons-le.

La pénalisation sur les performances supersoniques est donc importante et n'a plus été acceptée en France après 1975, date à laquelle il a pu être montré que la formule du MIRAGE 2000 permettait une vitesse d'approche voisine de 150 kt.

18. Les planches qui précèdent sont d'une présentation très schématisée, les problèmes comme la stabilité ou les performances en combat aérien sont bien sûr plus complexes et ont nécessité de longues études en soufflerie, sur ordinateurs ou simulateurs. Mais il est souvent bien agréable et rassurant lorsque des résultats d'études complexes peuvent être présentés par des raisonnements simples.

Un autre conférencier a présenté un schéma triangulaire montrant les trois rôles d'un avion de combat (planche 9) :

1) - Défense aérienne ou interception

2) - Supériorité aérienne

3) - Attaque au sol.

C'est ainsi que disposant d'un moteur donné, le M53 de 9 tonnes, nous pensons avoir optimisé la formule aérodynamique autour des missions d'interception et de supériorité aérienne en abandonnant à nouveau l'empennage arrière.

Et quant au troisième sommet du triangle, le MIRAGE 2000 sera également un excellent avion d'attaque au sol, comme l'a été le MIRAGE 3.

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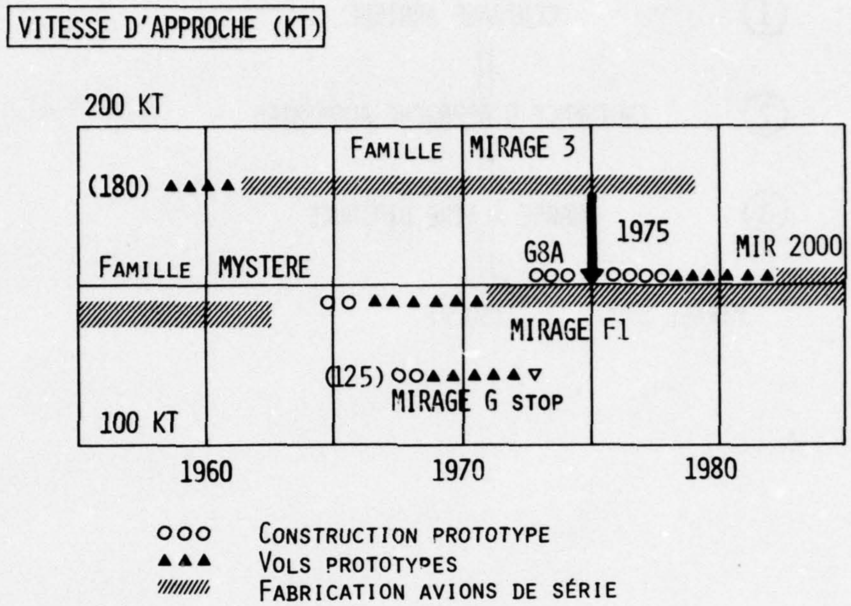


PLANCHE 1

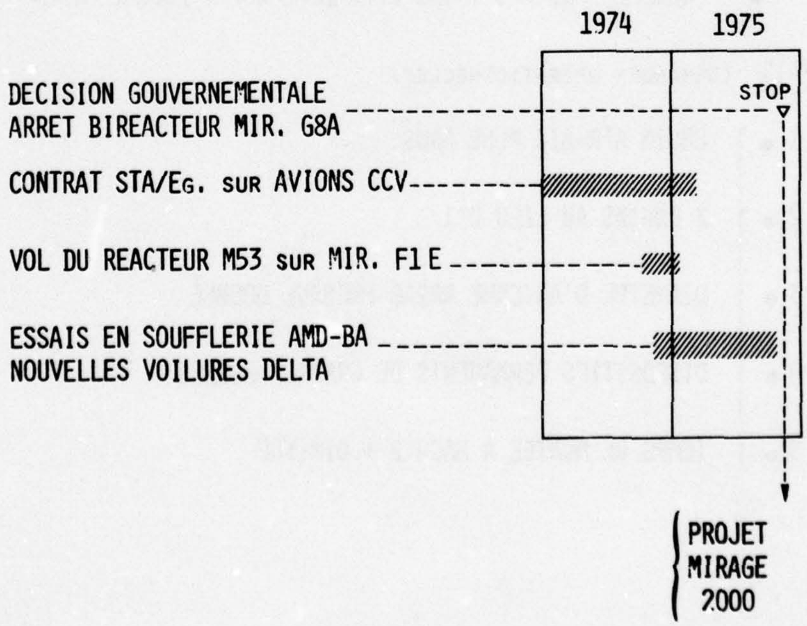


PLANCHE 2

MIRAGE 3 — 180 KT

↓

1. CENTRAGE ARRIERE

2. INCIDENCE D'APPROCHE AUGMENTEE

3. CHARGE ALAIRE DIMINUEE

↓

MIRAGE 2000 — 150 KT

PLANCHE 3

COMPARAISON PROGRAMMES MIRAGE 2000 / MIRAGE 3

- POUSSEE M53 = 9 T (AU LIEU DE 6 À 7 T POUR L'ATAR)

MAIS (DEMANDES OPERATIONNELLES)

1 • ENGIN AIR-AIR PLUS GROS

2 • 2 ENGINS AU LIEU D'1

3 • DIAMETRE D'ANTENNE RADAR PRESQUE DOUBLÉ

4 • DISPOSITIFS PERMANENTS DE CONTRE-MESURES

5 • TEMPS DE MONTEE A MACH 2 + DIMINUÉ

PLANCHE 4

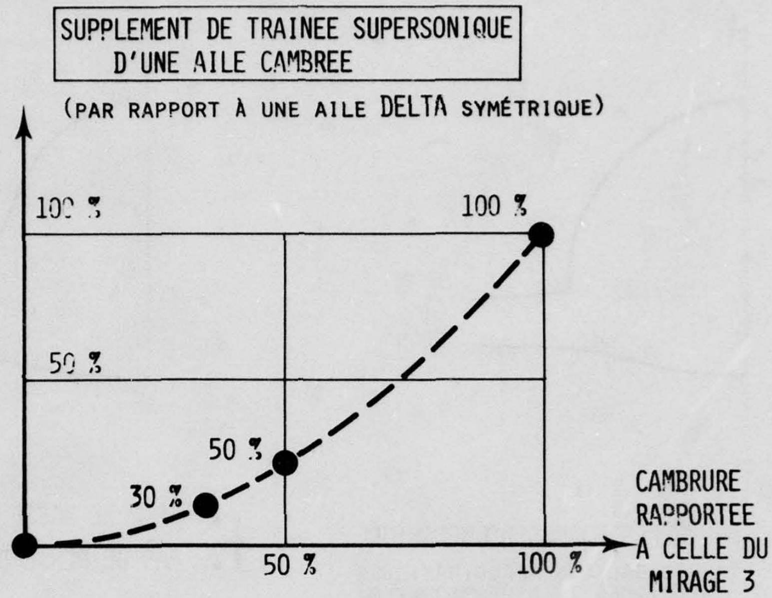


PLANCHE 5

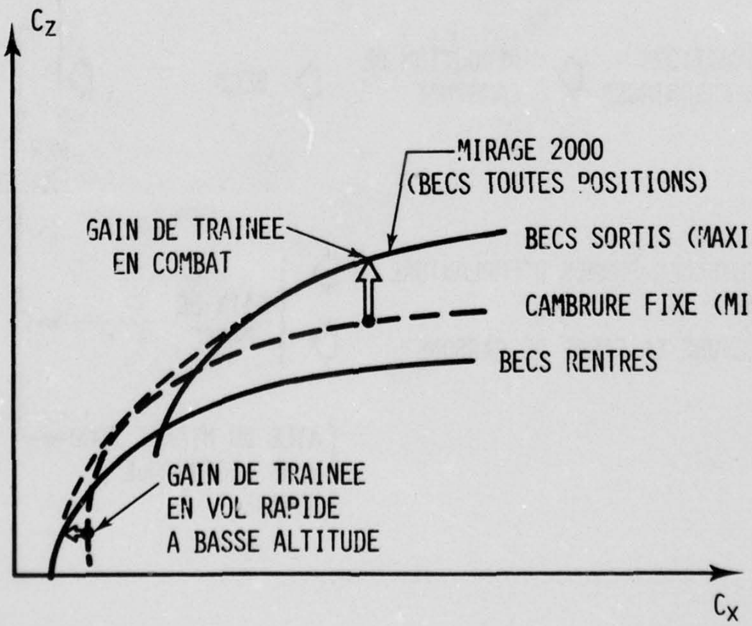
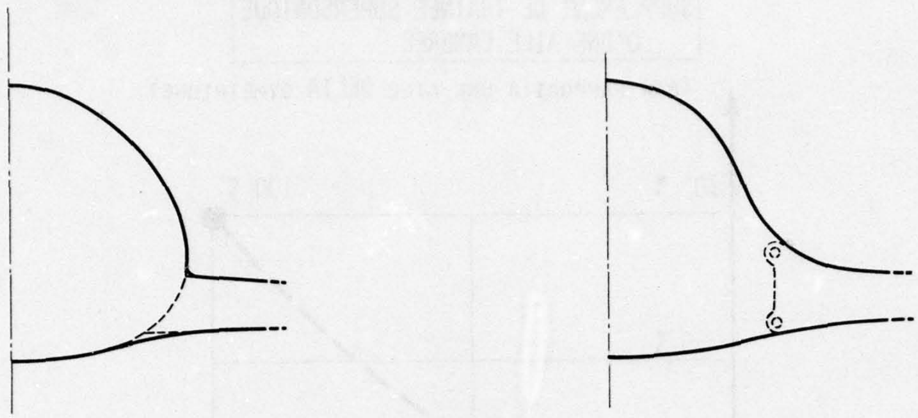


PLANCHE 6

MIRAGE 3 MIRAGE 2000



NOUVELLES FORMES D'EMPLANTURE = { • GAIN DE MASSE • PAS DE PERTE DE TRAINEE }

(APRES CALCULS AERODYNAMIQUES THÉORIQUES TRIDIMENSIONNELS)

PLANCHE 7

150 KT ⇨ SURFACE AILE AUGMENTEE (∼20 %) ⇨

CONDITIONS SUPERSONIQUES ⇨ REDUCTION DE CAMBRURE ⇨ BECS ⇨

→ PERTE DE MASSE

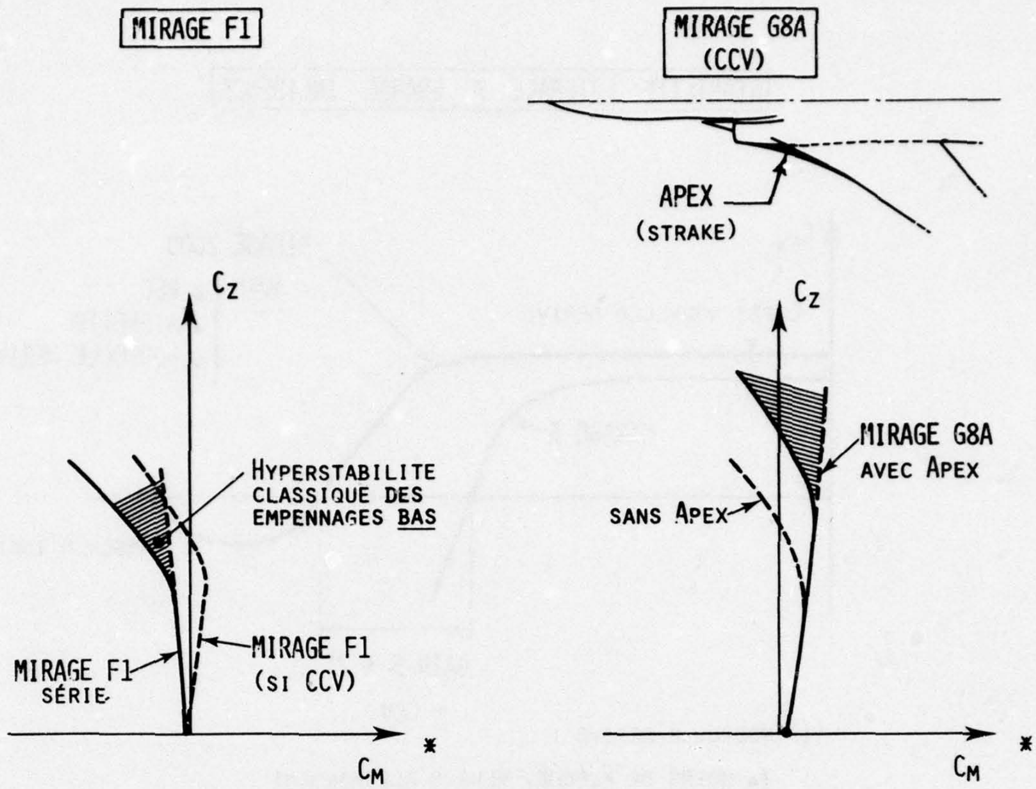
NOUVELLES FORMES D'EMPLANTURE ⇨

ELEVONS EN FIBRE DE CARBONE ⇨

→ GAIN DE MASSE

→ [AILE DU MIRAGE 2000 PLUS LEGERE QUE MIRAGE 3]

PLANCHE 8



* CONVENTION DE SIGNE INVERSE AUX U.S.

PLANCHE 9

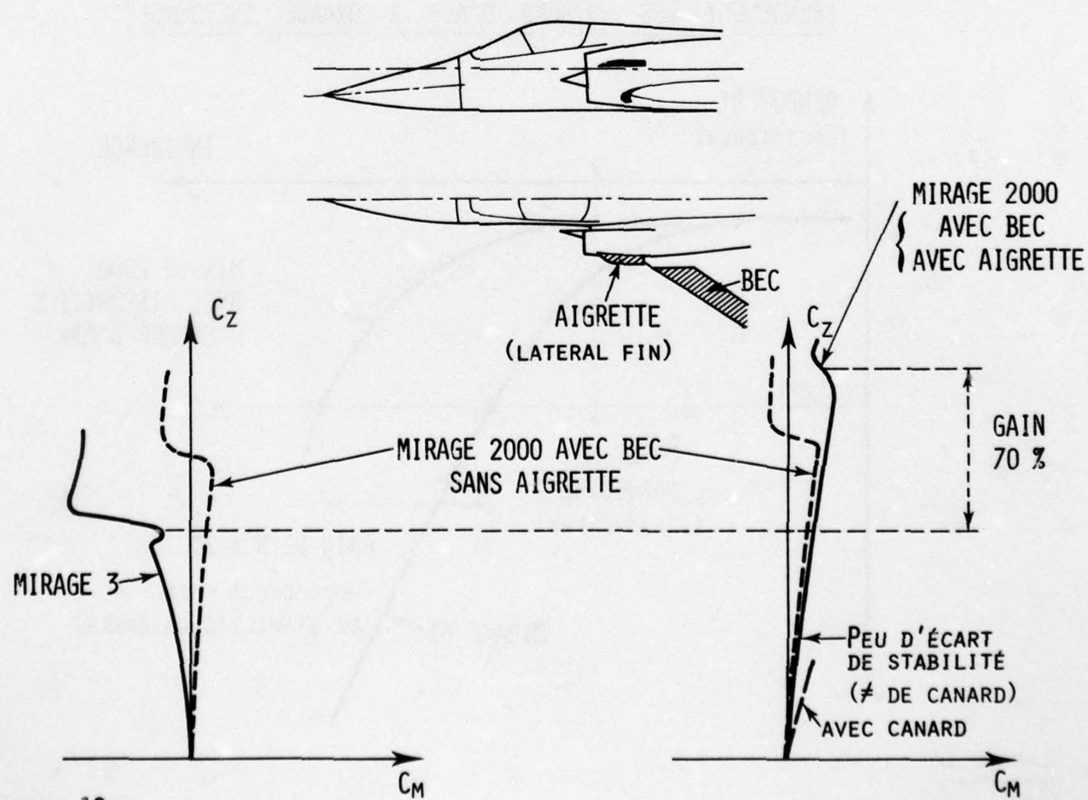
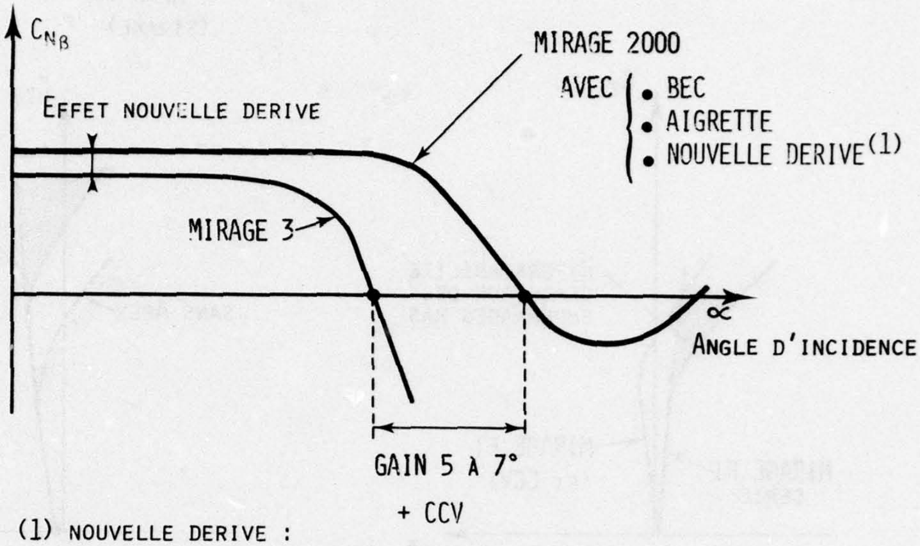


PLANCHE 10

STABILITE LATERALE A GRANDE INCIDENCE



(1) NOUVELLE DERIVE :

- MOINS DE FLÈCHE, PLUS D'ALLONGEMENT
- CONSTRUITE EN FIBRE DE CARBONE (DRAPEAU: MIXTE BORE/CARBONE)

PLANCHE 11

RENDEMENT DES ENTREES D'AIR A GRANDE INCIDENCE

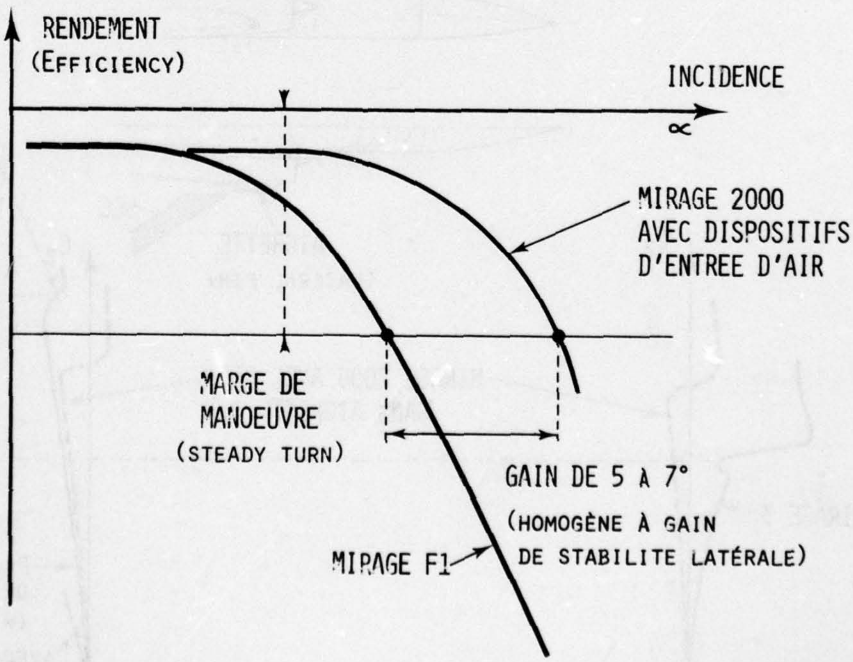
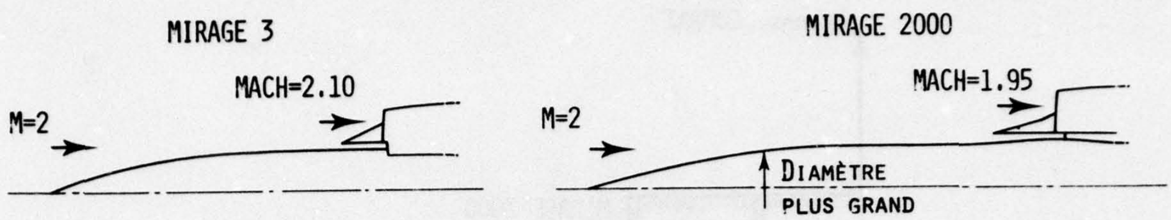


PLANCHE 12

AERODYNAMIQUE DE LA POINTE AVANT



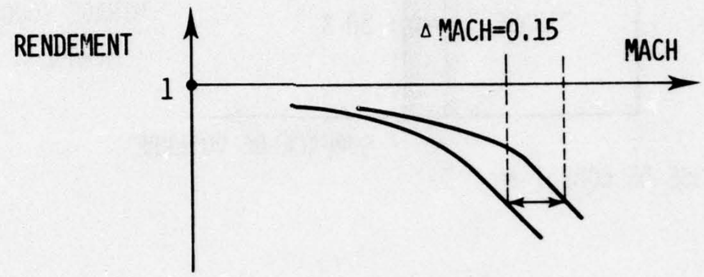


PLANCHE 13

MIRAGE 2000

AERODYNAMIQUE EXPERIMENTALE

ESSAIS CLASSIQUES ——→ PROLONGES JUSQU'A 35/40°
(TOUTES SOUFFLERIES FRANCAISES)

ESSAIS ENTREES D'AIR MACH=0.9 α = 30°
ONERA (MODANE) (S1 - DIAMÈTRE 8 M)

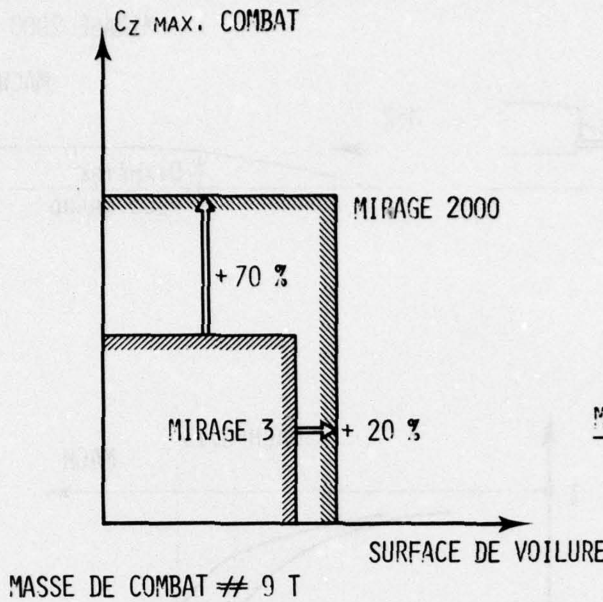
COEFFICIENTS DYNAMIQUES EN ROTATION A GRAND α.
IMFL (LILLE)

MAQUETTE DE VOL LIBRE (CATAPULTE)
(FREE FLIGHT MODEL) IMFL (LILLE)

EFFET DE SOL - CHARIOT SUR BASSIN HYDRAULIQUE
(GROUND EFFECT) CEAT (TOULOUSE)

PLANCHE 14

LIMITES DE MANOEUVRE (1)

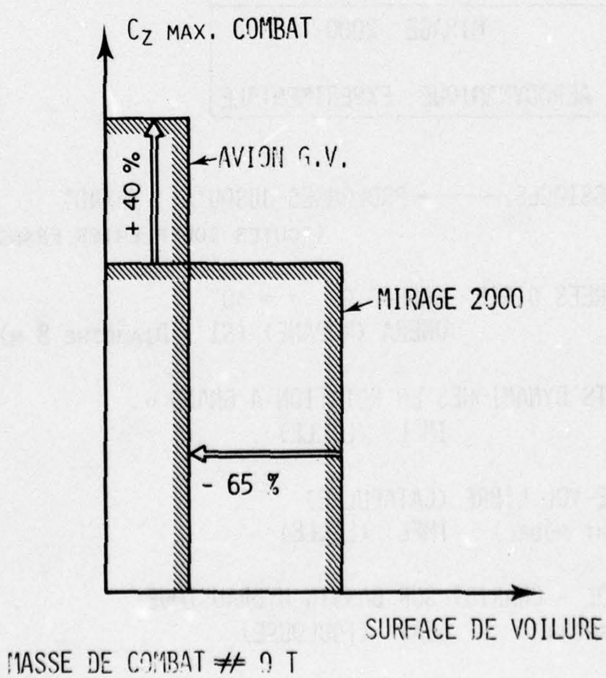


$$1.7 \times 1.2 = 2.0$$

$$\frac{\text{MIRAGE 2000}}{\text{MIRAGE 3}} = 2.0$$

PLANCHE 15

LIMITES DE MANOEUVRE (2)



$$1.4 \times 0.35 = 0.5$$

$$\frac{\text{MIRAGE 2000}}{\text{AVION G.V.}} = 2.0$$

⇩

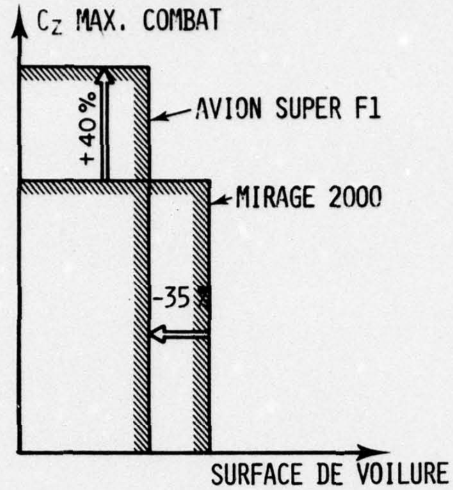
AVION G.V.
INCAPABLE
SUPÉRIORITÉ AÉRIENNE

⇩

ARRÊT AVIONS G.V.
EN FRANCE

PLANCHE 16

LIMITES DE MANOEUVRE (3)



$1.4 \times 0.65 = 0.91$

$$\frac{\text{MIRAGE 2000}}{\text{AVION SUPER F1}} = 1 + \varepsilon$$

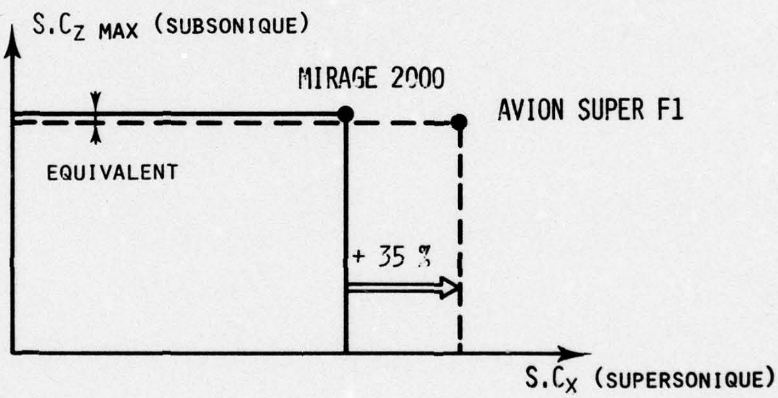
⇩

EXAMEN DU SUPERSONIQUE POUR DEPARTAGER LES 2 FORMULES

MASSE DE COMBAT # 9 T

PLANCHE 17

COMPARAISON MIRAGE 2000 / AVION SUPER F1



DONC ⇨ EN FRANCE, LA FORMULE D'AVIONS À EMPENNAGE ARRIÈRE EST ABANDONNÉE FIN 1975 POUR LES AVIONS DE COMBAT

PLANCHE 18

AVION DE COMBAT

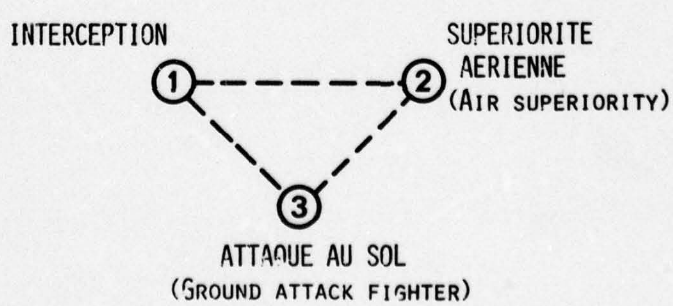


PLANCHE 19

SUPERCRUISER FIGHTER ANALYSIS

by
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Wright-Patterson Air Force Base, Ohio 45433

SUMMARY

A Supercruiser fighter is a system that cruises efficiently at supersonic speeds and is effective in air to air combat. The requirements lead to compromises in the fighter configuration. This paper will not address such compromises, but will focus on the supersonic cruise and maneuvering problems.

By maintaining a speed advantage over any threat aircraft, a Supercruiser can elect to engage or disengage at will thereby maintaining control over the combat arena. The most difficult situation occurs for a tail chase of the threat. It is shown that if the Supercruiser is twice as fast, then the combat radius equals twice the initial separation. The biggest improvement in supersonic combat radius results from adding fuel.

Against current fighters, a gun is effective whenever the target attempts to escape. From differential game technology, Supercruiser maneuvering requirements can be determined as a function of the threat's maneuvering capability.

Wind tunnel results indicate that a small single engine aircraft can be configured to produce the level of efficiency necessary for supersonic cruise. The realization of all performance goals in a single fixed geometry seems attainable.

INTRODUCTION

The optimal fighter is a weapon system that maintains control over the combat arena. Its probability of engaging, killing, and surviving the enemies' fighters should be near one. Ideally it can enter and leave the arena at will. While these characteristics are obviously desirable, they are most unlikely to appear in one fighter system. Some existing fighters are outstanding in the arena but they have trouble engaging the enemy. This is primarily a result of the mission radius dropping off rapidly as they go supersonic. Thus, new systems are confronted with the question relative to the importance of supersonic cruise performance versus transonic maneuvering requirements. New fighters must have satisfactory transonic performance if they are required to provide air cover for ground strike aircraft.

The pros and cons have waged several years on the merit of speed advantage. Proponents argue that closure is faster and escape more likely if attacked by an enemy. The opponents contend that too much speed leads to an overshoot with the attacker becoming the pursued. Both these arguments are valid. This paper will attempt to provide some insight into this problem.

The way that the problem will be approached is to first determine the requirements for reaching the combat arena. This will be called the intercept analysis. The arena is qualitatively defined as the volume in space where the friendly and enemy systems can engage each other. For situations where the engagement does occur, then the outcome will be evaluated. Speed will be the primary variable in the analysis.

INTERCEPT ANALYSIS

Capture is defined as those conditions for which one aircraft, called the attacker, can employ its armament against the other fighter, hereafter called the target. The target is attempting to return to its base and the attacker seeks to prevent this. The optimal trajectories are constant heading paths. The relative headings can vary from tail to nose attacks depending upon the geometry of the trajectories.

If the range to capture is large relative to the weapon envelope, then it can be assumed for the kinematics of the intercept analysis that point capture occurs. If the envelope is large relative to the cruise range, then either the armament is long range or the cruise range is quite small. This is not the situation that we are interested in, at least, for this study.

Two dimensional geometry suffices for this analysis. Figure 1 illustrates the constant heading trajectories where $\sigma_A(\sigma_T)$ are the attacker (target) heading angle. R_o is the initial separation at which the chase starts and λ is the line-of-sight angle. Corresponding speeds are V_A and V_T. β is the angle-off measured relative to the attacker's velocity. For point intercepts, it can be easily shown that

$$V_A \sin(\lambda-\sigma_A) = V_T \sin(\lambda-\sigma_T) \quad (1)$$

$$\dot{R} = V_T \cos(\lambda-\sigma_T) - V_A \cos(\lambda-\sigma_A) \quad (2)$$

Equation (1) states that the rate of rotation of the line-of-sight angle (λ) is zero. Equation (2) is the closing rate ($\dot{R}$) between attacker and target and is constant if the speeds are constant. The time to capture T_c is

$$T_c = -\frac{R_o}{\dot{R}} \tag{3}$$

Capture requires sufficient fuel for flight times greater than T_c since the attacker must return to its base or a refueling point if tankers are employed.

If the geometry is such that a tail attack occurs, then

$$\sigma_A = \sigma_T = \lambda$$

and

$$\dot{R} = V_T - V_A$$

For nose attacks

$$\sigma_A = \lambda$$

$$\sigma_T = \sigma_A + \pi$$

$$\dot{R} = -V_R - V_A$$

The flight time for tail attacks is larger than for nose attacks.

The attacker's cruise range (R_{AC}) is related to the heading angles and R_o.

$$R_{AC} = V_A T_c$$

$$= \frac{V_A R_o}{V_A \cos(\lambda - \sigma_A) - V_T \cos(\lambda - \sigma_T)}$$

$$= \frac{R_o}{\cos\beta_A - u\cos\beta_T} \tag{4}$$

where

$$u = V_T/V_A$$

$$\beta = \lambda - \sigma$$

Hereafter we will use β_A and β_T since the relative geometry is important. If the target is faster than the attacker, then u is greater than one and for capture to occur it follows that

$$\cos\beta_A > u\cos\beta_T \tag{5}$$

The significance of this relation is illustrated in Figure 2. The circles at A and T are of radii V_A and V_T respectively. For the attacker heading illustrated there are two heading solutions for the target. V_T along T1 corresponds to the paths diverging. The intercept solution occurs only along T2′. Thus if $u>1$, intercept can occur only for nose hemisphere attacks, i.e. β_T greater than 90°.

The attacker has little if any control over the value of the target heading. Thus β_T should be treated as an independent parameter for the capture analysis. In Figure 3 the necessary condition for intercept, Equation (1) is presented as a function of β_T and V_T/V_A. These results are necessary but not sufficient for intercept. Note that two values of β_T (except 90°) yield the same solution for β_A. This follows from the argument used in Figure 2. If the attacker is faster, $u<1$ intercept can be achieved for any β_T. This is not the case if the target is faster. Also, there are two solutions for β_A, but $\beta_A \leq 90°$ yields shorter capture times and cruise range. There exists values of β_T for which capture could not occur. As an example, if $u = 2.0$ and $\beta_T = 45°$

or 135°, no intercept can occur. A special case is $V_T = V_A$, $\beta_T = 90°$. From Figure 3 it can be seen that $\beta_A = 90°$ which means the attacker and evader fly parallel to each and therefore never intercept if R_o is greater than zero. This is obviously a trivial case.

Whereas Figure 3 depicted necessary conditions for intercept, Figure 4 along with Figure 3 presents necessary and sufficient conditions. As an example of what this means consider $u = 1.5$, $\beta_T = 30°$. From Figure 3 a selection exists for β_A, namely 48.6°. Figure 4 shows that this combination falls in the no capture area. The explanation is that $\dot{R}$ is positive so the target flies away from the attacker. Also, for an intercept, it is necessary that β_A be less than or equal to 90°. The boundary in Figure 4 corresponds to $\beta_A = 90°$. This boundary delineates the possible and impossible capture conditions. Nondimensional capture time $V_T T_C / R_o$ is presented in Figure 5.

Maximum values of V_T/V_p in terms of β_T are illustrated in Figure 6. This solution results from setting $\beta_A = 90°$ in Equation (1). This upper limit on V_T/V_A corresponds to situations where the evader's speed is equal to or greater than attacker's speed. If the attacker is faster, eventual capture is always insured for all values of β_T.

From Figure 4 it is evident that for $\beta_T \geq 90°$ there exists an optimum value of V_A that minimizes R_{AC}/R_o. Solving Equation (1) for β_A, substituting into Equation (4), and differentiating with respect to u gives the optimal value for u(u*) minimum R_{AC}/R_o (R^*_{AC}/R_o), and optimal $\beta_A(\beta_A{}^*)$

$$u^* = -\cot\beta_T \tag{5a}$$

$$\frac{R_{AC}{}^*}{R_o} = \sin\beta_T \tag{5b}$$

$$\beta_A{}^* = \beta_T - \frac{\pi}{2} \tag{5c}$$

These values are presented in Figure 6. The physical explanation for this is quite simple. The solution is that the attacker's velocity is perpendicular to the target's velocity. The situation is illustrated in Figure 7. For the same value of R_o, the attacker flies the minimum distance if $V_A = V_A{}^*$. The solution is the minimum distance to the evader's path. Consequently for nose hemisphere attacks there is an optimal V, $V_A{}^*$.

If time to intercept is minimized, a different solution results. Minimum time corresponds to maximum closing rate which is minimum $\dot{R}$ in Equation 3. This means that the component of V_A along the line-of-sight vector is a maximum subject to satisfying the constraint for intercept. In Figure 2 it can be seen that as V_A increases, its component along R_o increases, and T_c decreases. As V_A increases, however, it eventually reaches a speed above which the combat radius decreases. Thus there is a tradeoff between time and cruise distance to intercept.

It is necessary to determine the sensitivity of T_c and R_{AC} due to changes in V_A. Since

$$T_c = \frac{R_{AC}}{V_A}$$

$$\ln T_c = \ln R_{AC} - \ln V_A$$

Differentiation gives

$$\frac{1}{T_c}\frac{\partial T_c}{\partial V_A} = \frac{1}{R_{AC}}\frac{\partial R_{AC}}{\partial V_A} - \frac{1}{V_A}$$

$$< \frac{1}{R_{AC}}\frac{\partial R_{AC}}{\partial V_A} \tag{6}$$

The left side is negative, thus T_c experiences a bigger change than does R_{AC} for equal changes in V_A. In other words, the intercept time is more sensitive to changes in V_A. The normalized time to intercept relationship is

$$\frac{1}{T_c}\frac{\partial T_c}{\partial V_A} = -\frac{1}{V_T}\frac{u}{\cos\beta_A(\cos\beta_A - u\cos\beta_T)}$$

The sensitivity parameter $\frac{V_T}{T_c}\frac{\partial T_c}{\partial V_A}$ is presented in Figure 8. The data show that at low values of u (attacker much faster than the evader) the intercept time is relatively insensitive to small changes in V_A. For increasing u the results are very sensitive to β_T.

Thus far, we have shown that if the attacker is faster than the target, then the intercept time and cruise range are less. Low values of target angle off yield largest time and range to intercept.

Nothing has been said about R_o other than it is the initial separation range for the intercept analysis. If the attacker operates autonomously, then R_o is its radar detection range. If the attacker operates in conjunction with ground or airborne control systems, then R_o is a function of the detection and tracking capability of the control system. This latter concept is scenario dependent. One way of handling this possibility is to assume that the target is detected as it crosses some boundary like the forward edge of the battle area (FEBA) or some line parallel to the FEBA. The object is to intercept the target as soon as possible. This minimizes the penetration by the target.

It is assumed that the attacker is at a distance L from the reference line and the target crosses the line perpendicularly. The attacker immediately assumes a collision course to intercept. The geometry is illustrated in Figure 9. The object is to minimize y for a given x. The target's angle-off is the supplement of that used earlier. Previously a pure pursuit tail chase gave $\beta_T = 0$, now $\beta_T = 180°$. Three independent equations can be solved simultaneously for y. The equations are

$$y = L - x\cot(\beta_A + \lambda)$$

$$\tan\lambda = \frac{x}{L}$$

$$\sin\beta_A = u\sin\beta_T$$

There are three unknowns β_A, β_T, y in these equations. Eliminating β_A and β_T yields

$$(1-u^2)z^2 + 2u^2z - u^2(1+w^2) = 0 \tag{7}$$

where

$$z = \frac{y}{L}$$

$$w = \frac{x}{L}$$

The conic sections described by Equation (7) are a function of the magnitude of u. The following table defines the section in terms of u.

u	Equation (7)
<1	Hyperbola
=1	Parabola
>1	Ellipse

The nondimensional penetration distance as a function of u and nondimensional offset distance is illustrated in Figure 10. As the attacker's speed increases, the target's penetration distance decreases. Attacker speeds less than the target's speed could probably handle this situation.

A review of the analysis conducted thus far shows that a need for a supersonic cruise fighter exists. Although it is not required for situations like those depicted in Figure 10, it is necessary if the supercruiser is to control the combat arena. From Figures 3 and 4, speed in excess of the target's speed was necessary for small target angles, i.e. tail chase situations. If the attacker can force the target to leave the combat area then the attacker is controlling the air space and it is unimportant whether or not the attacker intercepts the target. If this is the case then supersonic cruise speed is not required. But what if the target returns after the attacker leaves. Then the attacker does not control the airspace unless a continuous stream of attackers are present which is hardly feasible. Thus to control the airspace, the attacker must intercept and destroy the target. He must be able to do this even if the target attempts to escape. There of course will be situations for which the attacker cannot

intercept the target. The next problem is to determine the volume of airspace within which intercept of the target can occur. How the attacker detects the target is important; however, for our analysis we will not be concerned with this detail and it will be assumed that detection has occurred.

At detection the attacker accelerates to the supersonic cruise speed. The target's speed is constant but may be supersonic. A tail chase then ensues since this is the hardest task for the attacker in terms of intercept cruise range. The target's speed if it changes does so instantaneously. This clearly favors the target. The target in order to escape must tradeoff increasing speed and decreasing cruise range. Thus there will be situations for which the target cannot change speed. At times, depending upon the scenario, he will be able to fly faster than the attacker. The attacker can not intercept the target if this occurs. The intercept problem is illustrated in Figure 11. As before, the engagement starts at the separation R_o and intercept occurs at R_{AC}. The ground rules for the attacker are: accelerate to cruise speed, cruise until intercept, make one pass at the target, and return to the initial point. Thus the radius of action is R_{AC}. Neglecting altitude differences, we see that the area the attacker controls consists of all points in the circle of radius R_o. This implies that an initial turn uses negligible fuel, distance, and time. This assumption is adequate for this simplified analysis.

Both acceleration and cruise segments are important. Different objectives could be considered for trajectory optimization. Examples are minimum time to intercept, minimum fuel required to intercept, and maximum range all of which are important. This analysis will be limited to the minimum time problem. From a tactical standpoint, minimizing time results in less time for the enemy to accomplish its function.

The minimum time problem will be based upon energy state approximations. In this formulation altitude and flight path angle dynamics are ignored. The differential equations are

$$\frac{dx}{dt} = V \tag{8}$$

$$\frac{dE}{dt} = \frac{V(T-D)}{W} = P_s \tag{9}$$

$$\frac{dW}{dt} = -\dot{m} \tag{10}$$

where x, E, w are range, specific energy, and weight. T, D, P_s and $\dot{m}$ are thrust, drag, specific excess power, and fuel flow rate. Minimum time to intercept is the addition of minimum time to climb and accelerate to cruise speed plus the cruise time to the intercept point. The solution is the well known Rutowski minimum time trajectory for climb and acceleration.

The minimum time(t) is derived from Equation (9)

$$t = \int_{E_i}^{E_f} \frac{dE}{P_s} \tag{11}$$

where E_i and E_f are the initial and final specific energies. The fuel and distance are evaluated from Equations (8), (9), and (10)

$$x = \int_{E_i}^{E_f} \frac{dE}{P_s/V} \tag{12}$$

$$\Delta W = \int_{E_i}^{E_f} \frac{dE}{P_s/\dot{m}} \tag{13}$$

The climb schedule, altitude as a function of speed or vice versa, is obtained by

selecting the combination along a constant specific energy that gives maximum P_s. Studies conducted thus far have shown that for supersonic speeds P_s is approximately constant along the optimal trajectory. Also, the optimal trajectory in the supersonic domain can be approximated by constant dV/dh. Fuel flow rates are also nearly constant. Thus Equations (11) through (13) are integrable. Let V_i and V_f denote initial and final speeds, then

$$t = \frac{1}{P_s}\left[\frac{dh}{dV}\left(V_f - V_i\right) + \frac{1}{2g}\left(V_f^2 - V_i^2\right)\right] \tag{14}$$

$$x = \frac{1}{P_s}\left[\frac{1}{2}\,\frac{dh}{dV}\left(V_f^2 - V_i^2\right) + \frac{1}{3g}\left(V_f^3 - V_i^3\right)\right] \tag{15}$$

$$\Delta W = \frac{\dot{m}}{P_s}\left[\frac{dh}{dV}\left(V_f - V_i\right) + \frac{1}{2g}\left(V_f^2 - V_i^2\right)\right] \tag{16}$$

Nondimensional relations can be derived through the following substitutions

$$\xi = \frac{xgP_s}{V_f^3}$$

$$\tau = \frac{tgP_s}{V_f^2}$$

$$\phi = \frac{\Delta W g P_s}{V_f^2 \dot{m}}$$

$$w = V_i/V_f$$

The transformed relations are

$$\xi = \frac{g}{2V_f}\,\frac{dh}{dV}\,(1 - w^2) + \frac{1}{3}\,(1 - w^3)$$

$$\phi = \tau = \frac{g}{V_f}\,\frac{dh}{dV}\,(1 - w) + \frac{1}{2}\,(1 - w^2)$$

These solutions are presented in Figures 12 and 13 in terms of the speed ratio (w) and acceleration term (dV/dh). As w increases, V_i approaches V_f and all acceleration characteristics decrease. Also, increasing dV/dh gives increasing values for the performance variables.

For comparison purposes optimal trajectories were derived for a lightweight supersonic cruiser. The first trajectory was an acceleration from 10,000 feet Mach one to 35,000 feet Mach two. The second trajectory consisted of an acceleration from same initial conditions with a zoom climb at the end to 60,000 feet Mach two. Comparison of the optimal and approximate results are presented in Table 1.

For cruise the change in the specific energy is negligible and it is convenient to treat weight as the independent variable. Assuming that cruise is steady state, that is speed and engine throttle do not change, then from Equations (8) and (10)

$$\frac{dx}{dw} = -\frac{V}{\dot{m}}$$

$$= -\frac{V}{T \cdot SFC}$$

Substituting L=W, T=D, and integrating gives

$$X_{cr} = \int_{W_f}^{W_i} \frac{V}{SFC}\,\frac{L}{D}\cdot\frac{dW}{W}$$

where W_i, W_f are the initial, final cruise weights. The range factor (R_F) id defined as

$$R_F = \frac{V}{SFC}\ \frac{L}{D}$$

If R_F is constant with respect to W and generally this is a satisfactory approximation, then

$$X_{cr} = R_F \ln \frac{W_i}{W_f} \tag{17}$$

which is the Brequet formula. The cruise time is easily computed from

$$t_{cr} = \frac{X_{cr}}{V_f} \tag{18}$$

The maximum radius of action can now be determined. Refer to Figure 14. Segment 1-2 is the acceleration, 2-3 and 4-5 are cruise segments, and 3-4 contains turning performance and expending payload. Let x_{ij} denote the distance between point i and j and W_i the weight at point i. The combat radius is the distance between points 1 and 3 which is equal to the distance between 4 and 5. From previous results

$$x_{12} + R_F \ln \frac{W_2}{W_3} = R_F \ln \frac{W_4}{W_5} \tag{19}$$

$$W_2 = W_1 - \Delta W_A \qquad \Delta W_A = \text{acceleration fuel}$$

$$W_4 = W_3 - \Delta W_C \qquad \Delta W_C = \text{combat fuel plus payload}$$

Reserve fuel (ΔW_{RES}) is

$$\Delta W_{RES} = W_5 - OWE$$

where OWE is the operating weight empty. Two parameters are introduced

$$k = \frac{\Delta W_{RES}}{\Delta W_{FUEL}}$$

$$\lambda = \frac{OWE}{W_1} \qquad 1 - \lambda = \text{fuel fraction + payload fraction}$$

Typical values for λ range from 1/2 for bomber type aircraft to 3/4 for lightweight fighters. Expanding Equation (19) and substituting the weight relations gives the following approximate nondimensional weight relation

$$\frac{W_3}{W_1} = \tfrac{1}{2}\frac{\Delta W_C}{W_1} + [\lambda + k(1-\lambda)]^{\frac{1}{2}} \left[1 + \tfrac{1}{2}\left(\frac{x_{12}}{R_F} - \frac{\Delta W_A}{W_1}\right)\right] \tag{20}$$

The combat radius (R_{AC}) is therefore

$$R_{AC} = x_{12} + R_F \ln \frac{W_2}{W_3}$$

$$= x_{12} + R_F \ln \left(\frac{1 - \dfrac{\Delta W_A}{W_1}}{W_3/W_1}\right) \tag{21}$$

The acceleration performance is easily determined from Equations (15) and (16) or Figures 12 and 13. Turning performauce is also a straightforward calculation.

The combat radius is sensitive to the acceleration characteristics (P_s), the range factor (R_F), and the fuel fraction $(1-\lambda)$. The sensitivity coefficients are the change in R_{AC} due to a change in one of these characteristics. The sensitivity coefficients

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are presented in Table 2. Only those characteristics that affect the combat radius are presented. Given a baseline configuration, the sensitivity coefficient identify the characteristic which yields the biggest improvement in R_{AC}. As an example assume the following aircraft values.

W_1 = 12000 pounds ΔW_C = 600 pounds

R_F = 3000 nautical miles ΔW_A = 1200 pounds

P_s = 1000 ft/sec k = 0.10

FUEL = 3000 pounds

The combat radius for this configuration is approximately 132 nautical miles. For a one per cent increase in fuel by either adding fuel or replacing equipment with an equal weight in fuel, R_{AC} increases by 3 and 4 nautical miles respectively. A one per cent increase in R_F or P_S yields approximately one nautical mile improvement. Adding fuel yields the biggest improvement in combat radius.

From Equations (20) and (21) required R_F can be determined as a function of R_{AC}. Representative parameters are

$$\frac{x_{12}}{R_F} = 0 \qquad k = 0.10$$

$$\frac{\Delta W_A}{W_1} = 0.1 \qquad \lambda = 0.6,\ 0.75$$

$$\frac{\Delta W_C}{W_1} = 0.05$$

The values for λ represent the variation in current and future fighters. The solution of Equations (20) and (21) is presented in Figure 15. As indicated before, the results are sensitive to fuel fraction. Supercruiser configurations should fall between the two fuel fraction curves. Current fighters at supersonic speeds have range factors less than 2000 nautical miles. Supercruiser studies have indicated that R_F varies from 4000 at M=1.6 to 3200 at M=2.0. For 25% fuel fraction R_{AC} varies from 150 to 200 nautical miles. For 40% fuel fraction R_{AC} varies from 450 to 550 nautical miles. The message here is that any equipment that has questionable value should be replaced with fuel, if feasible to do so.

The intercept envelope can now be determined. Refer to Figure 11. The threat will be intercepted if the initial separation is less than or equal to R_o. It is assumed that a tail chase occurs, if not the initial separation can be larger. The acceleration of the threat is ignored. From Figure 11

$$R_o + V_T\, t_{AC} = R_{AC}$$

Therefore

$$R_o = R_{AC} - V_T\, t_{AC}$$

Since

$$t_{AC} = t_2 + \frac{R_{AC} - x_{12}}{V_f}$$

$$R_o = R_{AC}\left(1 - \frac{V_T}{V_f}\right) - \frac{V_T}{V_f}\left(V_f t_{12} - x_{12}\right)$$

Using typical values like those used before and t_{12} equals one minute, R_o can be computed as a function of speed ratio, fuel fraction, and range factor. The solution for R_o is presented in Figure 16. The initial separation decreases with increasing speed ratio. If the initial separation is greater than 175 nautical miles, then a 0.25 fuel fraction cruiser can not even reach a stationary target.

For application purposes R_o is scenario dependent. Given R_o, then data like that in Figure 16 identifies the range of values for fuel fraction and R_F that result in intercept. Given that an intercept can be achieved, the next problem is that of determining what type of weapon is required to destroy the target. This is the subject of the next section.

where W_i, W_f are the initial, final cruise weights. The range factor (R_F) id defined as

$$R_F = \frac{V}{SFC}\ \frac{L}{D}$$

If R_F is constant with respect to W and generally this is a satisfactory approximation, then

$$X_{cr} = R_F \ln \frac{W_i}{W_f} \tag{17}$$

which is the Brequet formula. The cruise time is easily computed from

$$t_{cr} = \frac{X_{cr}}{V_f} \tag{18}$$

The maximum radius of action can now be determined. Refer to Figure 14. Segment 1-2 is the acceleration, 2-3 and 4-5 are cruise segments, and 3-4 contains turning performance and expending payload. Let x_{ij} denote the distance between point i and j and W_i the weight at point i. The combat radius is the distance between points 1 and 3 which is equal to the distance between 4 and 5. From previous results

$$x_{12} + R_F \ln \frac{W_2}{W_3} = R_F \ln \frac{W_4}{W_5} \tag{19}$$

$$W_2 = W_1 - \Delta W_A \qquad \Delta W_A = \text{acceleration fuel}$$

$$W_4 = W_3 - \Delta W_C \qquad \Delta W_C = \text{combat fuel plus payload}$$

Reserve fuel (ΔW_{RES}) is

$$\Delta W_{RES} = W_5 - OWE$$

where OWE is the operating weight empty. Two parameters are introduced

$$k = \frac{\Delta W_{RES}}{\Delta W_{FUEL}}$$

$$\lambda = \frac{OWE}{W_1} \qquad 1 - \lambda = \text{fuel fraction + payload fraction}$$

Typical values for λ range from 1/2 for bomber type aircraft to 3/4 for lightweight fighters. Expanding Equation (19) and substituting the weight relations gives the following approximate nondimensional weight relation

$$\frac{W_3}{W_1} = \tfrac{1}{2}\frac{\Delta W_C}{W_1} + [\lambda + k(1-\lambda)]^{\frac{1}{2}}\left[1 + \tfrac{1}{2}\left(\frac{x_{12}}{R_F} - \frac{\Delta W_A}{W_1}\right)\right] \tag{20}$$

The combat radius (R_{AC}) is therefore

$$R_{AC} = x_{12} + R_F \ln \frac{W_2}{W_3}$$

$$= x_{12} + R_F \ln \left(\frac{1 - \frac{\Delta W_A}{W_1}}{W_3/W_1} \right) \tag{21}$$

The acceleration performance is easily determined from Equations (15) and (16) or Figures 12 and 13. Turning performance is also a straightforward calculation.

The combat radius is sensitive to the acceleration characteristics (P_s), the range factor (R_F), and the fuel fraction (1-λ). The sensitivity coefficients are the change in R_{AC} due to a change in one of these characteristics. The sensitivity coefficients

COMBAT ANALYSIS

The ability of a fighter to intercept another aircraft is a necessary condition, however it is not sufficient for killing it. The fighter must be able to reach a firing position from which its weapon can destroy the target. There are at least two ways for addressing the weapon problem. The first is to select a weapon and then evaluate it by means of an air-to-air combat simulation program. The problem with this approach is that it is difficult if not impossible to evaluate the effect of changes in weapon characteristics on the outcome. The second approach and the one that will be used here is to apply the results of differential games to define the type of weapon that should be employed. The advantage of this approach is that it defines, at least approximately, the relationship between outcomes and performance variables. The disadvantage is that the roles are structured, i.e. the objectives do not change during the course of the engagement.

The two aircraft engaged in combat will be the target and attacker as before. The weapon capability will be defined by separation distance and angle-off. The angle-off is the angular displacement between the attacker's heading and the line-of-sight vector from attacker to target. Zero angle-off corresponds to pure pursuit. For a gun the separation must be less than 3000 feet and the angle-off less than one degree. For missiles, the angle-off and separation are limited by the avionics, propulsion and aerodynamic capability. If the attacker cannot reach a gun-firing position, then a missile must be employed. We seek to define those regions in space where a missile must be used in lieu of a gun.

The attacker will be faster than the target, otherwise the target can escape. If the target attacks rather than tries to escape, then the engagement converts generally to nose to nose. The outcome is related to the weapon capability rather than the launch platform. This is an important problem but it will not be studied here.

Since the target is defensive, its best tactic is to fly straight and then make a hard turn. The timing for the turn is critical. It is assumed that the attacker has detected the target at such separations that initial maneuvers by the target do not improve its situation. The longitudinal accelerations are negligible thus the maneuvering by each aircraft is limited by the aerodynamic and structural constraints.

The primary performance variables are the speeds and minimum radius of turn. The attacker attempts to come as close as possible to the target subject to satisfying the angular constraint. Conversely the target attempts to maximize the miss distance. From report AFFDL TR-75-47, "Application of Differential Games to Air-to-Air Combat Problems", data can be derived for evaluating minimum miss distances. Data are presented in Figure 17 for an angle-off constraint of one degree. The miss distance is L and the minimum radii of turn are R_A and R_T. Clearly increasing V_A gives decreasing miss distance ratio. The aforementioned report also shows that miss distance is more sensitive to speed changes than other variables. With decreasing R_A/R_T, the miss distance eventually goes to zero. It is shown in the aforementioned report that the relationship for which the miss distance is zero is

$$\left(\frac{V_T}{V_A}\right)^2 \frac{R_A}{R_T} \leq 1 \qquad (22)$$

It is easily shown that this relation can be interpreted in terms of the normal accelerations. If the attacker's normal acceleration is equal to or greater than that for the target, then the miss distance is zero. For this condition and under optimal maneuvering by the attacker, the target's survival probability is likely to be small.

In Figure 18 sustained load factors for a preliminary Supercruiser vehicle and a fighter typical of current capability are presented. The current fighter's maximum structural load factor is 9. For discussion purposes assume that the Supercruiser's structural limit is 7. Based upon previous discussions, the Supercruiser can force the miss distance to zero down to the n=7 current fighter's contour. Everywhere above this contour the Supercruiser's load factor is higher. Below this contour the load factor varies from 7 to a maximum of 9. Therefore since the target's (current fighter) load factor is higher, the miss distance will not be zero. Data like that in Figure 17 can be used to determine the miss distance. Assume that below the limit contour, the Supercruiser's speed is on the maximum dynamic pressure limit. The altitude is the target's altitude. In Figure 19 is presented the miss distances for the n=8 and 9 contours. The angle-off constraint is one degree. The miss distances are within gun capabilities, therefore this Supercruiser configuration is a potent weapon system as long as it maintains supersonic speed.

It should not be inferred that a gun is the only weapon that a Supercruiser needs. The previous discussion was based upon the assumption that the target was trying to escape. The situation where the target is aggressive is a different problem. Air-to-air missiles are required here particularly if the target has missiles. Also, at supersonic closing rates there is little time to point the aircraft in the direction necessary for a gun kill.

In the next section aerodynamic requirements for a Supercruiser are discussed.

AERODYNAMIC REQUIREMENTS

The effectiveness of a Supercruiser is dependent upon its combat radius and maneuvering capability. The combat radius involves the proper blend of fuel fraction, engine performance, and aerodynamic characteristics. The maneuvering capability is a function of the latter two. The following discussion is based upon the selection of a generic engine.

Optimal cruise performance corresponds to maximum range factor (R_F)

$$R_F = \frac{V(L/D)}{SFC} \tag{23}$$

At a given speed, the mathematical maximum value for R_F would correspond to maximum L/D and minimum SFC. In the supersonic region this does not occur, which is illustrated in Figure 20. Ideally the three curves would coalesce. Along the altitude for minimum SFC, sustained cruise yields lift coefficients less than the optimum for maximum L/D. Consequently the altitudes are different depending upon which factor is optimized.

For cruise, L/D can be derived from Equation (23)

$$\frac{L}{D} = \frac{R_F SFC}{V}$$

For the generic engine, SFC varies from 1.2 to 1.6 at M=1.4 for max dry and min afterburner operations. At M=2.0 the variation is 1.6 to 2.2. In Figure 21 the required L/D is presented. The range factor varies from 3000 to 4000 nautical miles. This gives combat radius between 150 and 500 nautical miles dependent upon the fuel fraction. The NASA Supersonic Cruise Airplane Technology (SCAT) program produced a maximum L/D of approximately seven at M=2.0 for a Supercruiser configuration. The current capability points correspond to wind tunnel data on a similar configuration. The cruise L/D for this configuration is 0.98 and 0.85 of max L/D at M=1.4 and 2.0, respectively.

The aerodynamic requirements for maneuvering capability are dependent upon the required sustained load factor. From the definition of the load factor (n)

$$n = \frac{L}{W}$$

For sustained turning performance, previous relation can be rewritten as

$$n = \frac{L}{D}\left(\frac{T}{W}\right) \tag{24}$$

From Equation (22) it was deduced that the miss distance could be theoretically driven to zero if the attacker and target load factors were equal. If the target load factor was greater, then the miss distance was not driven to zero for all target speeds. Figure 19 illustrates this point. Given a target load factor capability like that in Figure 18, the required aerodynamic capability can be determined. Since Equation (24) corresponds to sustained attacker load factor, the target's load factor is used until the attacker's maximum load factor is reached. In other words, target load factors are used if they do not exceed maximum attacker load factor. If they do, maximum attacker load factor is used. Consequently minimum required aerodynamic capability, $(L/D)_{req}$, is

$$(L/D)_{req} = \frac{n}{T/W}$$

where

$$n = \begin{cases} n_T & n_T \le \bar{n} \\ \bar{n} & n_T > \bar{n} \end{cases}$$

where $\bar{n}$ is the attacker's maximum load factor. For Supercruiser engagements along the dynamic pressure limit or M=2.0, the requirements based upon the data in Figure 18 are:

M	h	$(L/D)_{max}$
1.2	0	5.0
1.6	15000	4.6
2.0	30000	4.2
2.0	40000	4.1
2.0	50000	4.2
2.0	60000	3.8

Supercruiser maximum load factor is 7.33. Based upon earlier discussion, cruise performance levies stringent requirements on the aerodynamic characteristics.

In the next section, aerodynamic concepts are addressed.

AIRCRAFT CONFIGURATION CONCEPTS

A set of aerodynamic characteristics were developed that allow an attacking aircraft to control the combat arena. Control implies the attacker must intercept and destroy the target. The attacker must be faster than the target, but high velocity must not be at the expense of extreme reductions in range. Supersonic range was shown to depend on high values of the lift-to-drag ratio. Maneuvers at high load factors should not generate low values of the lift-to-drag ratio either, or engine thrust will be too low to sustain the maneuver at constant speed. The search for a supercruiser then becomes the search for an aircraft configuration which achieves high values of the lift-to-drag ratio at both supersonic cruise and transonic maneuvering conditions. The performance studies showed that the cruise condition was the most demanding mission constraint.

An aircraft configuration was developed by Boeing for the Flight Dynamics Laboratory, to illustrate the aerodynamic capabilities of a highly loaded single engine supersonic cruise fighter. The minimization of cruise drag was achieved through selection of a highly swept arrow wing with a thin cambered airfoil using the technology developed by NASA in their supersonic cruise aircraft technology programs. The fuselage was cambered to match the wing and made as slender as pilot, equipment, and propulsion system constraints would allow to reduce the zero-lift drag coefficient. The configuration is shown in Figure 22. A model of the aircraft concept was built and tested at subsonic through supersonic speeds. The results of these wind tunnel experiments showed several characteristics which are unique and may be typical of this special class of aircraft.

Supercruise potential was the first item to be established. The drag data was adjusted to correct for the internal flow through the engine duct and corrected for the presence of the force balance and model support. Trim drag was also included. The skin friction correction from wind tunnel conditions to full scale was not included, since our experience has been that the decrease in drag due to scale is nearly offset by additive increments due to manufacturing roughness and excresence items. The zero-lift drag coefficient is shown in Figure 23(a).

The maximum lift-to-drag ratio exceeded the cruise performance requirements. At subsonic speeds the maximum trimmed value was greater than 10, and at the supersonic cruise Mach number the maximum lift-to-drag ratio was 5.4 shown in Figure 23(b). The trim condition noted was for cruise along the q=500 p.s.f. line. The parameter $M(L/D)_{max}$ was increasing for all supersonic Mach numbers, from 7.3 at Mach 1.05 to 11.6 at Mach 2.0.

Lines of constant lift-to-drag ratio were constructed on an altitude Mach number map for the cruise case, Figure 24(a), and the 7.3 load factor maneuvering case, Figure 24(b). The aerodynamic efficiency requirements presented for maneuvering in the previous section are exceeded at altitudes below 40,000 feet, but the data does not show the desired values at 50,000 and 60,000 feet.

The arrow planform and slender profile combine to produce a general set of aerodynamic parameters which are well behaved and suited for operation over a wide range of speed and load factor. The lift coefficient was essentially a linear function of angle of attack for the range of these tests. The variation in the lift curve slope with Mach number, Figure 25(a), was small. This characteristic has been observed in several highly swept configurations such as the X-24B flight test vehicle which had a 78° swept leading edge.

There was a large change in the static margin (dC_m/dC_L) between subsonic and supersonic speeds. The shift in aerodynamic center was 8% of the mean aerodynamic chord. This change will produce a very stable aircraft at supersonic speeds. Nevertheless, the aircraft exhibited adequate control characteristics.

The model exhibited a strong nose-up pitching moment at the zero-lift condition. The large positive pitching moment coefficient has several advantages in terms of flight characteristics. Rotation at take-off would occur without a negative deflection of the trailing edge elevators and the control deflections for flight trim at subsonic speeds increase the lift-to-drag ratio. At the supersonic cruise condition the aircraft is

nearly self trimming and the few degrees of elevator deflection required to produce the cruise lift coefficient cause a very small trim drag increment.

The longitudinal aerodynamics were the principal objective of the study, so longitudinal controls were the only ones addressed in the test program. On this model the trailing edge was deflected to represent elevator controls. A leading edge flap was also designed into the model and the effects of a maneuvering flap to reduce flow separation was measured at subsonic and transonic conditions.

The model was designed with fixed trailing edge flap angles of $\pm 7.5°$. At all Mach numbers the change in lift and pitching moment coefficient due to flap deflection, Figure 26, was quite linear. The elevator power is adequate at the transonic conditions, to pitch the aircraft to lift coefficients in excess of 1.0 with less than 10° of flap deflection. The change in drag coefficient was an order of magnitude less than the change in lift or a given deflection angle. This feature, coupled with the increase in lift with positive deflections, produces an improvement in lift-to-drag ratio when the trimmed case is considered.

The leading edge flap was most effective in modifying the pitching moment characteristics. The change in lift and drag was very small, although the direction of the change was to produce small increases in the lift-to-drag ratio at high lift conditions. The pitching moment coefficients, shown in Figure 25, were much more regular when the leading edge flap was deflected. Both 10° and 15° deflections are shown, and the effect shows little change between the two deflections. It appears that a variable deflection angle would not be required for this control. The flap would simply be deployed to 10° when transonic maneuvers were executed so that the overall vehicle control was more linear and regular.

CONCLUSIONS

Based upon the preceding analysis, the following results were obtained.

1. In order to control the combat arena, a speed advantage over the threat is required. For a speed ratio (V_T/V_A) equal to 0.5, the capture range for a tail chase is twice the initial separation range.

2. The biggest improvement in combat radius is through adding fuel. The volume for equipment of marginal value should be used for fuel.

3. Against a current fighter, a gun can be effective whenever it attempts to escape. From an aerodynamic design standpoint, a significant challenge is developing a configuration that has adequate supersonic cruise performance.

4. From differential game technology, Supercruiser maneuvering requirements can be determined as a function of the threat's maneuvering capability.

5. The wind tunnel test data establishes that a small single engine aircraft can be configured to produce the level of efficiency necessary for supersonic cruise.

6. High load factor sustained maneuvers at altitudes greater than 40,000 feet can only be accomplished by resorting to afterburner thrust, with an accompanying sacrifice in range due to high SFC and low L/D. The realization of all performance goals in a single fixed geometry configuration is still a subject for study, but the goal certainly seems attainable.

TABLE 1

	OPTIMAL	APPROXIMATE
Final Altitude (feet)	35,000	35,000
Distance (nautical miles)	11.9	13.6
Time (minutes)	0.9	0.9
Fuel (pounds)	530	550
Final Altitude	60,000	60,000
Distance	17.7	20.0
Time	1.3	1.2
Fuel	690	750

TABLE 2

R_F	λ	0.6	0.75
	SENSITIVITY TO FUEL FRACTION		
3000		2042	1691
4000		2723	2255

$\Delta W_A/W_1$	λ	0.6	0.75
	SENSITIVITY TO RANGE FACTOR		
0.05		0.166	0.073
0.10		0.137	0.044

R_F/P_s	$\Delta W_A/W_1$	0.05	0.10
	SENSITIVITY TO ACCELERATION		
3		0.083	0.180
4		0.111	0.241

R_F (nautical miles), P_s (feet/second)

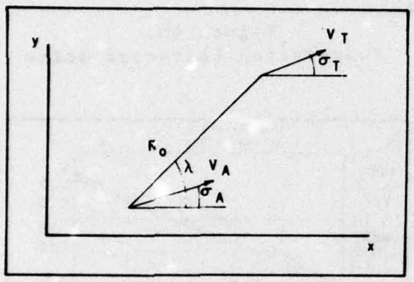

Figure 1.
Capture Geometry

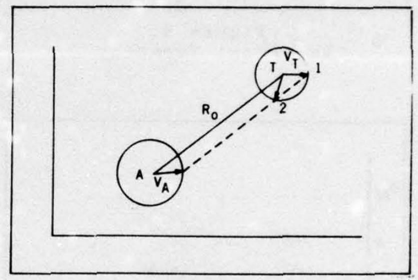

Figure 2.
Collision Course Geometries

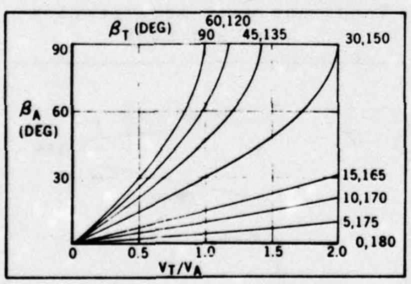

Figure 3.
Intercept Angle

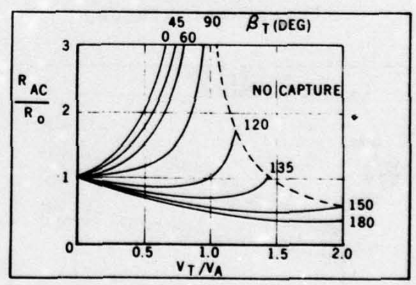

Figure 4.
Capture Range Ratio

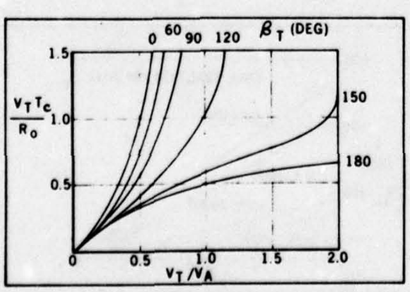

Figure 5.
Nondimensional Intercept Time

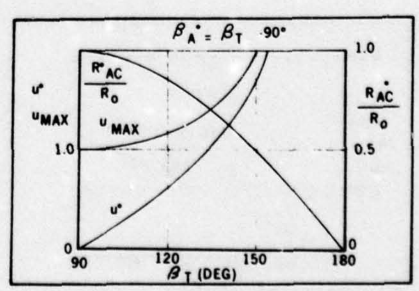

Figure 6.
Optimal Capture and Speed Ratios,
and Maximum Speed Ratio

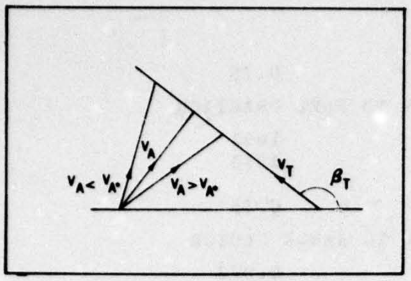

Figure 7.
Optimal Intercept Geometry

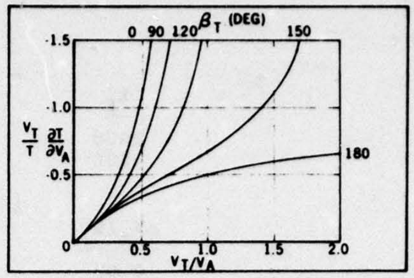

Figure 8.
Time to Intercept Sensitivity

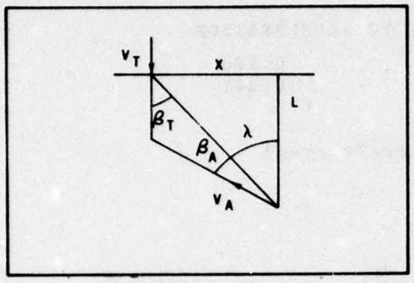

Figure 9.
Intercept Geometry

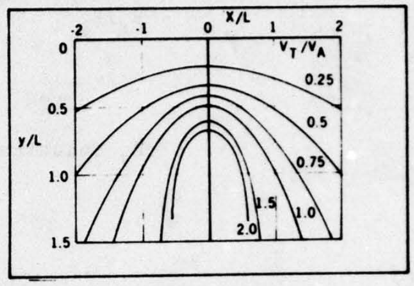

Figure 10.
Penetration Characteristics

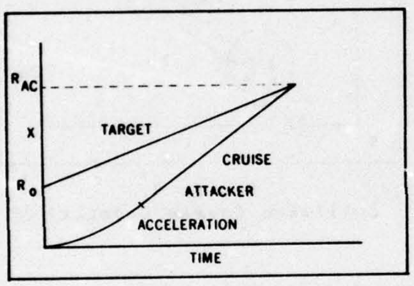

Figure 11.
Engagement History

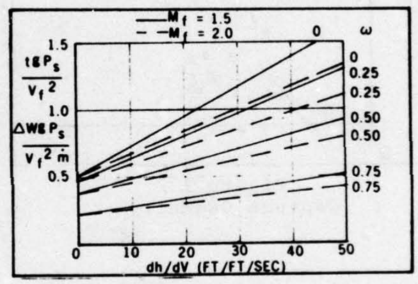

Figure 12.
Nondimensional Acceleration Time and Fuel

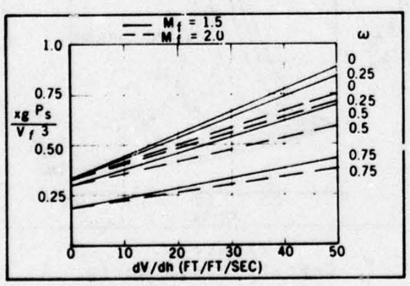

Figure 13.
Nondimensional Acceleration Distance

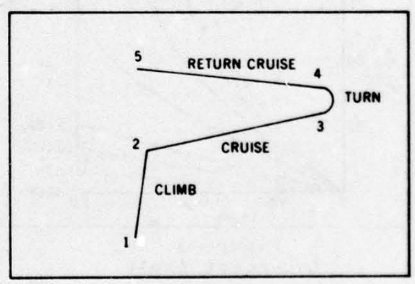

Figure 14.
Mission Profile

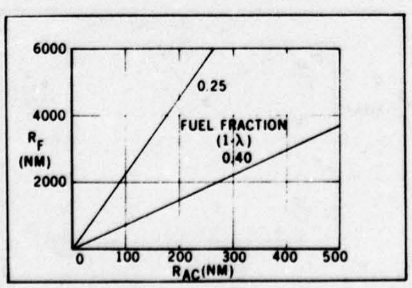

Figure 15.
Required Range Factor

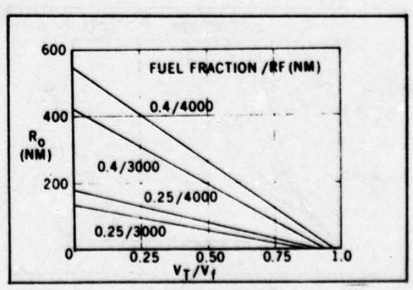

Figure 16.
Maximum Initial Separation For Intercept

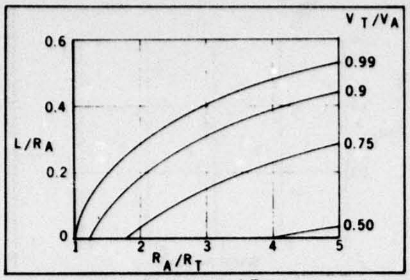

Figure 17.
Nondimensional Miss Distance

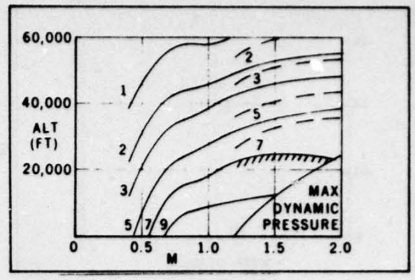

Figure 18.
Sustained Load Factor

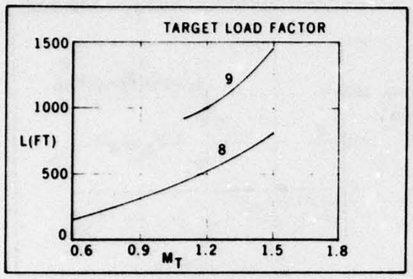

Figure 19.
Miss Distance

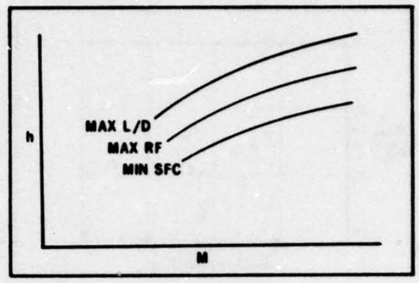

Figure 20.
Optimum Performance

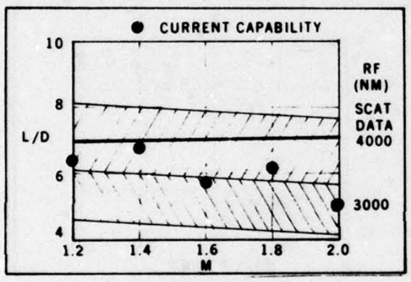

Figure 21.
Required Aerodynamic Cruise Capability

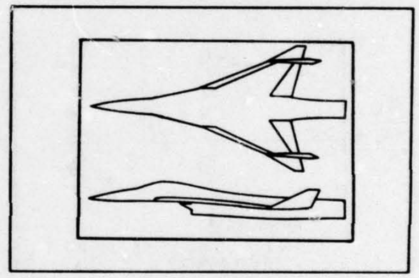

Figure 22.
Supersonic Cruise Fighter Concept

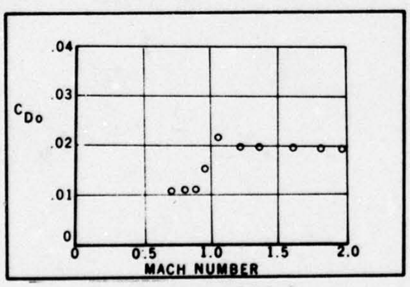

Figure 23(a).
Zero - Lift Drag Coefficient

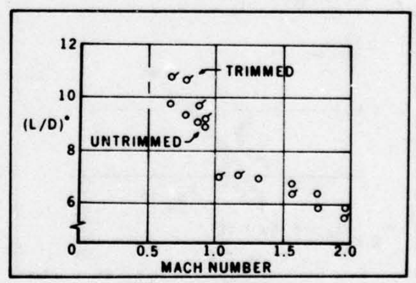

Figure 23(b).
Maximum Lift - to - Drag Ratio

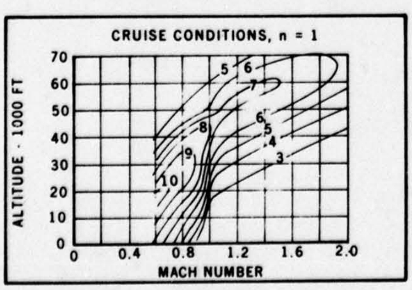

Figure 24(a).
Lift - to - Drag Ratio Characteristics
(Cruise)

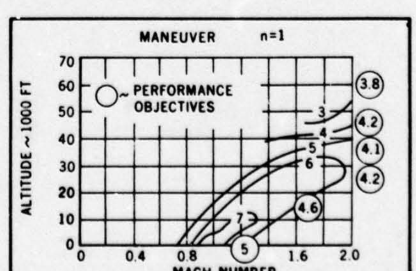

Figure 24(b).
Lift - to - Drag Ratio Characteristics
(7.3 G Maneuver)

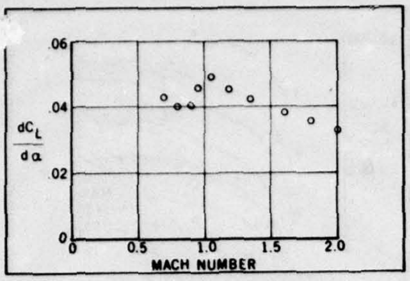

Figure 25(a).
Linearized Longitudinal Aerodynamic Characteristics (Lift Coefficient Slope)

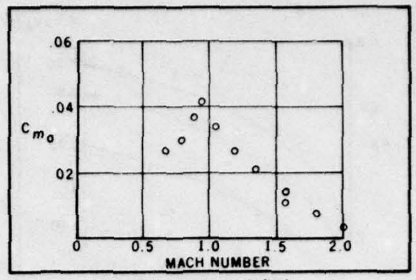

Figure 25(b).
Linearized Longitudinal Aerodynamic Characteristics (Zero - Lift Pitching Moment Coefficient)

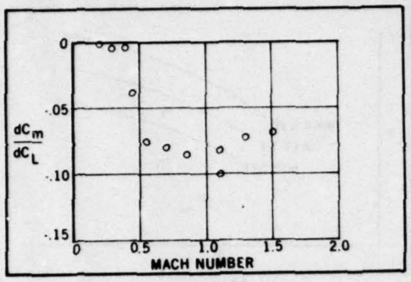

Figure 25(c).
Linearized Longitudinal Aerodynamic Characteristics (Static Margin)

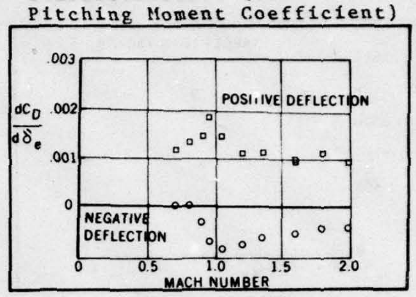

Figure 26(a).
Longitudinal Control Characteristics (Drag Derivative)

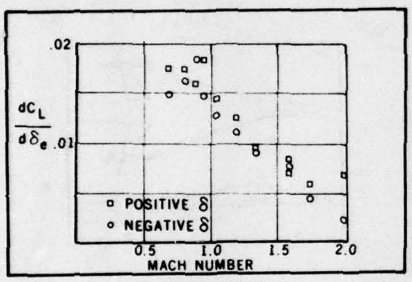

Figure 26(b).
Longitudinal Control Characteristics (Lift Derivative)

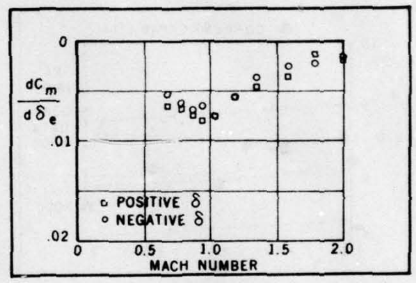

Figure 26(c).
Longitudinal Control Characteristics (Pitching Moment Derivative)

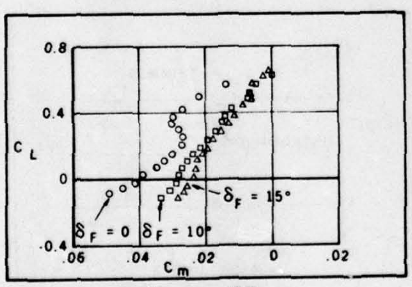

Figure 27.
Pitching Moment Characteristics of Leading Edge Flap

ANALYSIS OF ADVANCED VARIABLE CAMBER CONCEPTS

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Summary

This paper presents a survey of variable camber devices used on contemporary fighter aircraft to improve maneuverability in the air combat flight regime. Some new variable camber concepts which offer potential for even greater benefits on future fighter aircraft designs are discussed. Finally some analysis is presented which provides some insight into the advantages that can be achieved in various "off design" conditions with the use of these variable camber concepts.

Introduction

Ever since Horatio Frederick Phillips, reference 1, filed a patent for an airfoil demonstrating superior lift capability due to drooping the leading edge, the aircraft designer has been struggling with the problem of optimizing the airfoil design to operate efficiently throughout the flight envelope. The concept of varying camber in flight to alter the lift characteristics is not new. The Wright Brothers employed variable camber through wing warping to obtain lateral control of the first successful airplane. Increasing the camber of the wing to achieve increased lift for take-off and landing has been common practice since the 1920's. However, the concept of improving the "off design" performance, particularly during the midmission phases of flight, is a more recent application of this technology. With the design of very thin wings for supersonic flight (c.f., A-3J, F-8U3), it was found that subsonic cruise performance could be enhanced through partial deflection of the variable camber leading edge devices used in landing. Subsequently, continued efforts to increase the maneuvering performance of combat aircraft have lead to numerous variable camber concepts specifically oriented to this goal. It is this application of variable camber, namely increased transonic maneuverability and off-design performance that will be addressed in this paper.

Variable camber devices have taken many forms over the years. Concepts have varied from plain flaps on the leading and trailing edges of the wing, to the more sophisticated leading edge slat designs. Recent advances in materials technology and innovative mechanical design concepts have provided the capability of smoothly varying camber with no discontinuities in the wing surface. The present analysis is concerned only with devices that vary camber by deflecting the leading and trailing edges of the wing. It is beyond the scope of the present paper to consider concepts which may change the thickness of the wing, extendable strakes, variable sweep, or other variable geometry devices.

Presented herein is a review of the applications of variable camber to enhance maneuverability on contemporary aircraft. This is followed by a survey of the more recent variable camber concepts under study in the United States, as well as in other countries. While this survey is intended to be comprehensive, it is recognized that it is far from all inclusive. Furthermore, new concepts are constantly evolving and it is all but impossible to keep abreast of all the development. Finally, analyses have been conducted to determine the benefits obtainable through the application of variable camber, at least to first order. This analysis is intended to provide some general observations as to what increases in complexity and structural weight can be tolerated without negating the improved aerodynamic performance of the various devices.*

CURRENT APPLICATIONS OF VARIABLE CAMBER

In reviewing the applications of variable camber concepts to contemporary aircraft in order to improve maneuverability, two basic approaches become apparent. The first is to retrofit variable camber devices on an existing aircraft with a demonstrated deficiency, while the second is the development of a new aircraft which considers variable camber to be an integral part of the design from the onset. The latter approach usually results in automatic or semi-automatic operation of the variable camber as a function of flight condition.

Probably the best known recent application of variable camber to improve transonic maneuverability is that of the maneuvering slat on the F-4E aircraft. Very similar flaps, with improved low speed characteristics will also be retrofitted on the U.S. Marines Corps F-4J aircraft. The objectives of the F-4E leading edge slat application was to increase the angle of attack for buffet onset and consequently increase the maneuvering lift coefficient. As indicated in reference 2, these slats were the subject of extensive

*The authors wish to express their appreciation to Mr. Lynn Trobaugh of the David Taylor Naval Ship Research and Development Center for his assistance in modifying the mission analysis code and obtaining the performance data that is presented.

wind tunnel and flight test evaluation. The final configuration that evolved is shown in figure 1. Based on wind tunnel data presented in figure 2, this configuration did increase the buffet onset angle of attack from 7.0° to 10.6° with a corresponding increase in maneuver lift coefficient of approximately 0.2., at 0.8 Mach number. These wind tunnel results were confirmed during subsequent flight tests. In fact, the buffet onset angles-of-attack as identified by the pilot were approximately 8.5° and 13.5° for the basic wing and the slatted wing configurations, respectively. This increase over that determined from wind tunnel data is attributable to lower buffet intensity at onset, as well as a slow down of the progression of airflow separation across the wing as angle of attack increases for the slat configuration.

Later models of the F-14A aircraft will employ leading edge slats in conjunction with trailing edge flaps. Because of the automatic wing sweep features of the F-14A, the variable camber devices, particularly the trailing edge flaps, must be retracted prior to 0.9 Mach number. Consequently the F-14A variable camber is designed to improve maneuvering performance at the low to mid transonic speed range. A comparison of the optimum drag polars for the basic F-14A wing and with maneuvering slats is shown in figure 3, for 0.6 and 0.85 Mach number. The F-14A variable camber system is designed to be automatic, with deployment of the devices being a function of flight condition; therefore, the variable camber drag polar represents an optimum envelope of several configurations. As can be seen from the data, there is a cross over at approximately 0.85 lift coefficient at both Mach numbers, where the clean wing demonstrates greater aerodynamic efficiency, indicating the basic F-14A wing was designed to possess more than the minimum maneuver capability. This crossover is typical of slatted configurations. This points up one of the problems of retrofit applications of variable camber, that is, less than the maximum performance is obtained from the devices, for some flight conditions, and a penalty might have to be paid in other flight regimes because the basic wing was designed to have some maneuver capability.

The Northrup F-5E aircraft employs variable camber in the form of plain leading and trailing edge flaps. The F-5E application is termed semi-automatic in that the pilot selects a predetermined flap deflection as a function of flight condition. The flaps for a particular flight condition on the F-5E are selected by means of a unique thumb switch arrangement on the pilot's throttle lever. The F-5E also employs 8° trailing edge flap deflection for subsonic cruise flight. The drag polars comparing the characteristics of the F-5E clean wing, with those obtained with various flap settings is shown in figures 4 and 5 for 0.6 and 0.9 Mach number respectively.

These data from reference 4 reflect the auto-shift feature of the F-5E variable camber system. This feature automatically shifts the flaps at 200 KIAS from 24°/20° leading/trailing edge, to 18°/16° and again at 250 KIAS to 12°/8°. Above 550 KIAS, the flaps are retracted to zero degrees on both leading and trailing edge. As can be seen from the data, a 17% increase in lift at a representive drag coefficient of 0.2 is achieved at 0.6 Mach number; the corresponding improvement at 0.9 Mach number is 10.5%. It should be noted that the F-5E is characteristic of other aircraft, in that the maneuverability improvements due to variable camber are reduced significantly at 0.9 Mach number.

As indicated in reference 5, the YF-16 aircraft was designed almost from the onset to utilize variable camber. In this aircraft, the variable camber consists of automatically deflecting plain leading edge flaps as a function of flight condition.

The YF-16 drag polar for 0.8 Mach number is shown in figure 6. As indicated, the automatic variable camber increases sustained turn capability 18% over that obtained with the basic 64A204 section, which was designed to yield maximum L/D at cruise. An unexpected benefit of variable camber on the YF-16 was the increased directional stability at higher angles of attack. As shown in figure 7, the addition of the Leading Edge Extention (LEX) or strake causes the configuration to become directionally unstable at high angles-of-attack. However upon deflecting the leading edge flaps 25°, the aircraft maintained directional stability throughout the angle of attack range investigated. For this reason, the automatic variable camber devices have also been programmed to deflect as a function of angle of attack.

The U.S. Navy F-18 aircraft is also being designed to exploit variable camber for improved maneuverability. The F-18 system is to be automatic and follows a rather complex schedule as shown in figure 8 from reference 6. Figures 9, 10 and 11 present normalized drag polars (drag at zero lift subtracted) for the F-18 with various combinations of leading and trailing edge flap deflection at 0.6, 0.8 and 0.9 Mach number respectively. It can readily be seen that significant improvements in maneuverability accrue for representative maximum turn rate levels of drag, at 0.6 and 0.8 Mach number. However, as shown in figure 11, the improved maneuver capability at 0.9 Mach has been considerably reduced. In fact except at the very high drag levels, the basic wing is almost as efficient as the "optimum" wing. This significant decrease in improvement due to variable camber was also noted by the authors in reference 5 for the YF-16.

From this brief survey, it appears that fighter aircraft maneuver enhancement through the application of leading and trailing edge variable camber concepts have gained wide spread acceptance. The "off design" benefits to accrue from the use of variable camber are greater, if the concept is considered in the initial design phases of a new aircraft, rather than applied as an "add-on" later.

ADVANCED VARIABLE CAMBER CONCEPTS

Smooth Contour Variable Camber Concepts

Probably the most extensive study of the application of conformally varying camber to fighter/attack configurations to date was reported in reference 7. A detailed analytical and experimental investigation was performed to assess the potential of variable camber for improving combat performance. The basic aircraft chosen for the study was the U.S. Navy F-8. The F-8 is a single seat fighter designed in the 1950's with supersonic capability. The F-8 was chosen as the baseline aircraft because of its performance characteristics and because the wing on the F-8 could be readily replaced with a variable camber wing for flight demonstration, which was an original long term objective of the program.

A new advanced technology wing (ATW) was designed as the basic variable camber wing. The wing planform was basically the same as the F-8 wing. However, the airfoil sections used for the new wing were of the thin supercritical type, figure 12, with a "peaky" pressure distribution. To achieve variable camber, the wing was desinged to have smoothly variable sections between 5% and 25% chord and between 72.5% and 90% chord. With these airfoil sections, the new wing had, as might be expected, very good cruise and loiter characteristics.

An experimental program was conducted on a 1/10 scale modelof the F-8 both with the basic and advanced technology wing. For the test, 6 leading edge and 5 trailing edge conformal flap deflections were fabricated. In addition, plain hinged leading and trailing edge flaps were fabricated and tested on the advanced technology wing for comparison. Wind tunnel tests were conducted in the mach range from 0.2 to 2.0 at the NASA Ames Research Center.

Based on the experimental data and existing F-8 data, performance comparisons were made between the basic F-8 with and without leading edge droop and the advanced technology wing with both plain and conformal leading and trailing edge flaps. As one would expect, the newly designed advanced technology wing shows significant performance improvements over the basic F-8 wing at subsonic free stream mach numbers even without leading and trailing edge flaps, either conformal or plain. The interesting comparison is the one between the advanced wing with plain flaps and with conformal variable camber, figure 13 and 14. In this case, at a C_L of 0.8 and mach numbers of 0.7 and 0.9 the conformal camber shows respective decreases in drag of 18% and 6%. In addition to reduced drag, the maximum useable lift coefficient is greater for the advanced technology wing with conformal camber than with plain flaps for every mach number tested, see figure 15. Maximum useable lift was defined by taking an increment in angle-of-attack above buffet onset which was determined by force and moment variation and trailing edge pressure.

At supersonic freestream mach numbers, a condition closer to that for which F-8 wing was designed, the advanced technology wing has a higher drag coefficient at every lift coefficient, figure 16. For instance, at a mach number of 2.0, the zero lift drag is 13% less for the basic F-8 wing than for the advanced technology wing. This should be expected for the supercritical type airfoil with a rather large leading edge radius used for the advanced technology wing. Neither the conformal nor plain flap can be used to advantage for this flight condition, although a slight negative deflection of the plain flap improves the drag slightly. The higher drag for supersonic mach numbers should not necessarily reflect negatively on either conformal variable camber or plain flaps. However, if the larger leading edge radius becomes necessary to the concept and if the aircraft mission contains a sizeable supersonic dash, then the advantages at cruise, loiter, and maneuver conditions would have to be traded off against the supersonic penalty.

Performance comparisons in reference 7 for the advanced technology wing with conformal and plain flaps are what one would expect from the data. The lower drag for the conformal flaps results in better performance during cruise and loiter and better combat fuel flow at a given maneuvering load factor. The higher maximum lift coefficient permits operations at a higher load factor. The investigation of reference 7 also included a comparison of the maneuvering performance based on wind tunnel data for the advanced technology wing with conformal flaps and a model of the F-4 aircraft with maneuvering slats. Load factor limits due to heavy buffet are essentially the same for the two concepts. However, because the advanced technology wing reaches these limits at lower drag, it shows 20% advantage in combat fuel flow at these conditions. In fact, the combat fuel flow is essentially the same for the F-4 with slats as for the basic F-4 at the same "g" level. It should be noted that the data used for the slats in this comparison represent a retrofit onto an existing wing and may not be an optimum utilization of maneuvering slats. In fact, it is not clear in the report whether or not trailing edge flaps were used in conjunction with the leading edge slats. Reference 8 shows a significant increment in buffet C_L for maneuvering slats in conjunction with trailing edge flaps over slats alone for the F-4 aircraft, figure 17. Also figure 18 from reference 9 shows that at combat maneuvering mach numbers and lift coefficients leading edge slats produce a significant decrease in drag over a plain wing. In this case at $C_L = 0.8$ there is a drag reduction of approximately 24% with the slats. Therefore, in this case, slats would show a significant saving in combat fuel flow for maneuvering flight.

A conformal variable camber concept, like any variable geometry system, will have design complexities that must be considered in the evaluation of the feasibility and potential for flight applications. The previously discussed investigation devoted considerable attention to development of mechanisms to achieve the required conformal deflections. The development and performance of another innovative mechanism to achieve conformal leading edge geometry changes have been reported in references 10 and 11. The concept, called "RAEVAM", can primarily provide small down and up nose droop for increasing

lift at moderate and high subsonic Mach numbers, respectively, thereby providing higher maximum lift coefficients and better maneuver performance, see figure 19. Larger droop angles can be achieved for landing and take off conditions. The mechanism operates quite successfully and there appear to be no obvious barriers to developing a flight worthy system.

I might be worthwhile to mention one concern that was referred to earlier in this paper in reference to applications to fighter aircraft. The airfoil shapes that this concept has been applied to are relatively thick with a rather large leading edge radius. These airfoils perform well in the subsonic flight regime. However this concept has no provision for modifying the leading edge radius. Therefore, if there are restrictions that prohibit the application of the concept to thin airfoils with reasonably sharp leading edges or if the concept does not show similar benefits for sharper leading edges, then its feasibility for fighter applications which almost surely will require a supersonic capability would appear to be questionable. Only further investigations and development of the concept will provide answers to these questions.

Segmented Variable Camber

A unique variable camber concept developed by the Vought Corporation is shown in figure 20. In this concept, the leading and trailing edge flaps are divided into four segments, with the hinge line skewed, i.e., not aligned with a constant chord element. The result is a wing which displays basically conical camber, with the capability to vary the camber of the leading and trailing edges independently through a wide variation of shapes. This concept has been tested extensively in the wind tunnel by the NASA, reference 12 and an analysis of the potential benefits of the application of the system on a conceptual fighter aircraft are reported in reference 13. Figures 21 and 22 present the drag polars for various leading and trailing edge deflections at 0.6 and 0.9 Mach number respectively. In these figures, the configuration LO/TO is the basic uncambered wing section. As can be seen, considerable improvement in maneuverability can be achieved at 0.6 Mach number, through optimum scheduling of the wing camber. However, the benefits appear to diminish significantly at 0.9 Mach number.

The analysis indicates that the same camber schedule which yields the minimum drag, also results in the highest buffet free lift coefficient. This is not the case at 0.9 Mach number. In fact, the data indicate that the highest buffet free lift coefficient can only be achieved through sequential scheduling the leading edge flaps and then at a lift coefficient of approximately 0.3, start scheduling the trailing edge flaps, with the result that the angle of an attack will decrease as the buffet free lift coefficient increases. Based on the wind tunnel results, this is not the schedule to achieve maximum lift to drag ratios.

A firm conclusion of the reference 13 study is that even with a weight penalty of 1000 lbs. required to implement the variable camber concept, buffet free sustained load factors 50 to 60 percent greater than those obtained for aircraft configurations with an uncambered or a fixed camber wing are achievable. The study further concludes that the principal advantage of this variable camber concept at 0.9 Mach number is the improvement in wing buffet onset lift coefficient, but at the expense of increased aircraft drag. This concept does have the additional flexibility that with the skewed hinge lines variable camber and spanwise twist distribution are both achieved simultaneously. The previous concepts had no provision for spanwise variation of camber.

Self Optimizing Wing Concept

A somewhat different application of conformal variable camber has been investigated at General Dynamics/Convair Division under sponsorship of the U.S. Navy and Air Force. The goal of the investigation is to develop a flexible wing/digital computer wind tunnel model system that will automatically optimize the wing configuration for a specified set of flight conditions. In operation the digital computer is loaded with a gradient projection optimization code which takes force and moment data from the wind tunnel balance system and predicts model configuration changes to optimize the desired aerodynamic parameters. The model is built with flexible panels and internal actuators which permit changes in camber, twist, leading edge radius and thickness distributions while maintaining a smooth wing contour. The computer commands, through the control system, changes to the wing geometry based on the optimization predictions to optimize the desired parameters within given constraints, see figure 23.

The objective of the concept is to give the designer the ability to automatically optimize wing geometries in the wind tunnel in a shorter time and at less cost than by conventional testing methods. Hopefully, also, the final configuration would be more optimum than that obtained from a finite number of "best guess" configurations. To verify the concept, a two-dimensional self-optimizing model airfoil was designed and built. The model was designed with five variable parameters: conformal leading edge deflection, conformal trailing edge deflection, leading edge radius, lower surface thickness distribution, and lower surface aft camber. The model had a one foot chord and one foot span and was constructed of steel including the flexible leading and trailing edge panels. Target airfoil shapes ranged from a NACA0010 to supercritical shapes, figure 24. Internal hydraulic actuators to vary the shape parameters were controlled by a servo system coupled to a digital/analog computer system.

The model and control system were tested both subsonically and transonically in General Dynamics/Convair wind tunnel facilities. The results were reported in references

14 and 15. Parametric runs were made in which the model was systematically articulated to verify model controllability and to supply a data base for comparing with the optimum airfoils. Figure 25 shows typical results for variation of trailing edge deflections holding other parameters constant. For optimization runs, a desired flight condition is specified with required constraints. For example, for cruise one might wish to minimize drag at a given lift coefficient or for maneuver one might wish to maximize lift coefficient at a given drag coefficient with a buffet constraint. Figure 26 is a typical result for a drag minimization optimization run at $M = 0.75$ and a lift coefficient, $C_L = 0.5$. The run converged satisfactorily to a drag coefficient lower than the minimum drag curve achieved through parametric variations of the model geometry. The two-dimensional tests showed that the self-optimization system was stable and convergent at both subsonic and transonic conditions. Because of tunnel wall effects and questions about flow two-dimensionality, it is difficult to assess the data quantitatively or to compare absolute drag values with other data.

The first phase of the self optimizing wing investigation showed enough potential that a second phase was undertaken under sponsorship of the Office of Naval Research, the U. S. Air Force Flight Dynamics Laboratory, and the U. S. Air Force Arnold Engineering Development Center. The purpose of the second phase of the study is to extend the concept to a three-dimensional model wing. This will permit a more realistic assessment of the concept and allow, at least in principle, application of spanwise variations in camber and twist. The three-dimensional wing operates essentially in the same manner as the two-dimensional airfoil. The number of variables in the optimization process of course increases.

A one-sixth scale semi-span model of a modern variable sweep fighter/bomber wing has been designed and fabricated. The design includes six pairs of internal actuators at each of two spanwise stations. The leading edge has deflection actuators at 15% and 25% chord. The trailing edge has deflection actuators at 65% and 80% chord. In addition there are a pair of leading edge sharpness actuators and a pair of actuators to modify upper surface thickness.

The model was tested in August 1977 at the Arnold Engineering Development Center at Mach numbers from 0.6 to 0.95. Detailed evaluation and analysis of the data is currently underway. Successful optimizations were achieved for a number of flight conditions. One interesting aspect of the optimizations was that in every case spanwise camber distributions were important. The feasibility of this concept will largely be determined by the pay-off in better design and wind tunnel cost savings compared to the added complexity and cost of the model and control system.

MISSION PERFORMANCE ANALYSIS

It is difficult to accurately assess the relative performance improvements of different variable camber concepts that have been developed for different wings and tested under different conditions. However, to obtain performance data sufficient to make some general observations about the potential of the different concepts, a mission performance analysis program has been used which incorporates the available experimental aerodynamic data for the concepts previously reviewed. Plain leading and trailing edge flaps, conformal leading and trailing edge flaps, leading edge slats, and the segmented leading and trailing edge flaps are considered in the analysis.

A fighter aircraft mission typical of a U. S. Navy fleet defense mission was used. One concession made in the fighter mission was to eliminate supersonic dash from the mission profile because of a lack of supersonic data. Since none of the concepts provide significant performance improvements supersonically, this compromise should not affect the relative merits of any concept. The mission includes warm-up, take-off, climb, cruise, loiter, combat, return cruise, landing and reserve segments. The combat mission segment consists of six maximum sustained "g" turns at a mach number of 0.9.

The aircraft design used was typical of current advanced fighter configurations and engines. The aircraft design had a gross take-off weight of 33,830 pounds and a combat thrust-to-weight ratio of 1.05. The basic wing chosen for the analysis was the basic advanced technology wing from reference 7. The aerodynamic data for the conformal and plain flaps were taken from reference 7 and therefore is completely consistant with the basic wing. The data for slats were obtained from references 9 and 17. Trimmed lift and drag increments were obtained and added to the basic wing. For this case the basic wing was used except during the combat phase of the mission; the assumption being that the slats are retractable without a drag penalty at all other times. The segmented flap data was obtained from reference 12. Here also lift and drag increments were obtained after adjusting for scale, trim, and aspect ratio and added to the basic wing.

It is obvious from the survey that each concept offers aerodynamic advantages over the basic fixed camber wing. In application, this improved performance is obtained at a cost of additional complexity and quite possibly additional wing structural weight. The purpose of this performance analysis was to investigate the impact of additional wing structural weight on the mission performance benefits realized from improved aerodynamics. The questions of survivability, maintenance, and reliability are not addressed here.

Initially, the basic wing was run in the mission analysis program. At the same gross weight and fuel load, the plain flap, conformal flaps, leading edge slat and segmented flap configurations were used for the same mission using the optimum drag polars for each case. Next the gross weight was held fixed and the assumption was made that increases in

wing weight due to the variable camber concepts are accounted for by reducing the fuel load. The mission analysis is redone for different increments in fuel load for each concept.

Figure 27 shows the change in mission range for each concept for increases in wing weight. The plain flaps, slats, conformal flaps, and segmented flaps have the same range as the basic wing with increases in wing weight of 7.0, 8.5, 17.0 and 30.0 percent respectively. Consequently a 10.0 percent increase in wing weight for the conformal flap reduces its range performance to that of the plain flap. Likewise the segmented flap performance is equivalent to the plain flap with a 23.0 percent increase in wing weight.

The rather large increase in range performance of the segmented flap concept is derived primarily from the cruise and loiter segments of the mission. It can be seen from figure 13 that the conformal camber does not given any significant improvements in drag below a lift coefficient of 0.4. The segmented flap on the other hand, as seen in figure 21, gives sizeable increments in drag improvement for the lower lift coefficients. Thus it is not surprising that the segmented flap shows improved loiter performance.

A word of caution should be noted relative to these performance results. The segmented flap data was obtained for a wing with conventional uncambered NACA airfoil sections (root-NACA65A005; tip-65A004), see reference 12. This data was incrementally added to the basic advanced technology supercritical wing for the performance analysis. This assumes that similar flap deflections would produce the same drag reductions for the supercritical wing at the low speed loiter conditions. If it does not, this would result in the range for the segmented flap concept being inflated somewhat.

Table 1 lists some of the combat performance parameters for the different concepts. It should be noted that with the type of combat mission chosen, maximum lift coefficient as determined by heavy buffet is not reached by any of the concepts. There are not drastic differences in maximum lift due to heavy buffet for the different leading edge devices, therefore the relative combat performance should not be significantly different if the combat segment was based on this parameter. The conformal and the segmented flap concepts have essentially the same combat performance and are somewhat better than any of the other concepts.

CONCLUSIONS

Based on the foregoing discussion and analysis, the following conclusions are drawn regarding the utilization of variable camber concepts to enhance fighter aircraft "off design" performance.

A. Variable camber devices have been successfully employed to improve the transonic maneuvering characteristics of several current fighter airplanes. Those aircraft where variable camber was an "add on" to an existing design did not realize as great a performance improvement as did those aircraft where variable camber was included in the design from the onset.

B. Both existing and advanced variable camber concepts yield the greatest improvements in maneuverability in the low to mid Mach number range, i.e., below M = .8, with the performance gains decreasing rapidly at M = .9.

C. Most of the data available for conformal variable camber concepts has been obtained for wings with relatively large leading edge radii. For fighter applications, unless the wing sweep is such that the leading edge remains subsonic, this would result in a substantial supersonic drag penalty. Further investigations should be done to demonstrate applicability to thin wings with relatively sharp leading edges.

D. Comparitive performance analysis of several advanced variable camber concepts has, as would be expected, shown that the choice of a particular concept is very sensitive to the design mission requirements. The "off design" performance due to variable camber can have a major impact on overall mission performance. The segmented flap concept in this study appears to offer the most versitility and allow optimization at more "off design" conditions. The ability to vary the spanwise distribution of camber be very important in advanced concepts.

TABLE 1 COMPARISON OF COMBAT PARAMETERS

	ATW & CF	ATW & PF	ATW & SLATS	ATW & SF
Combat Time (T/T_{ATW})	0.878	0.908	0.916	.878
Load Factor ($L.F./L.F._{ATW}$)	1.113	1.097	1.089	1.135
Combat C_L	.7420	.7189	.7146	.7476

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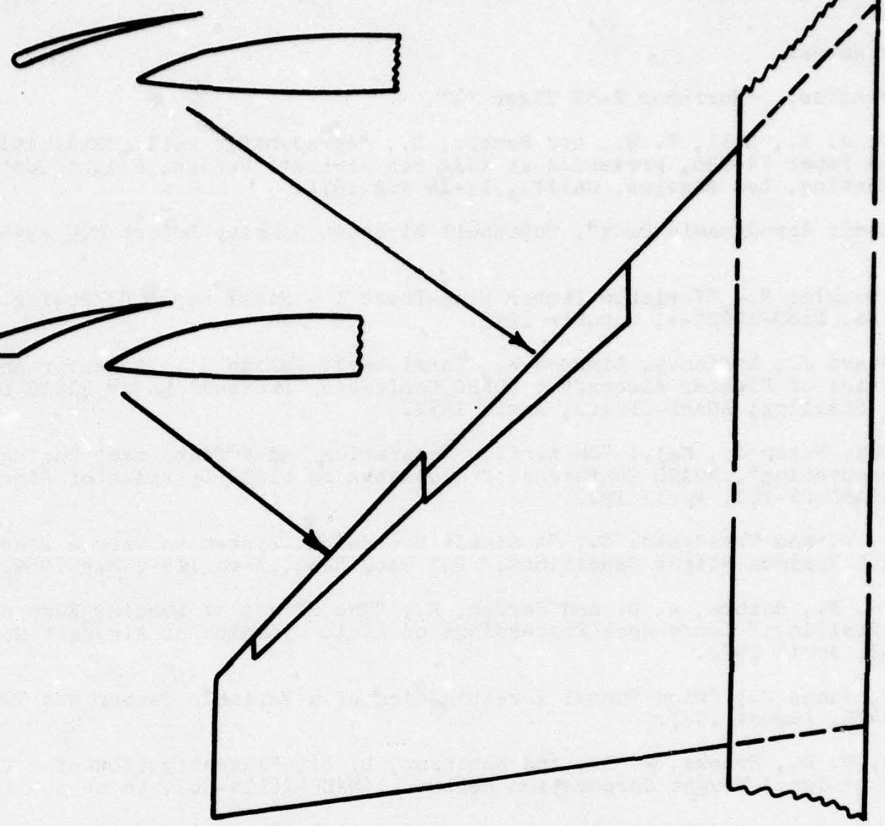

Fig.1 F-4E Maneuvering slat configuration

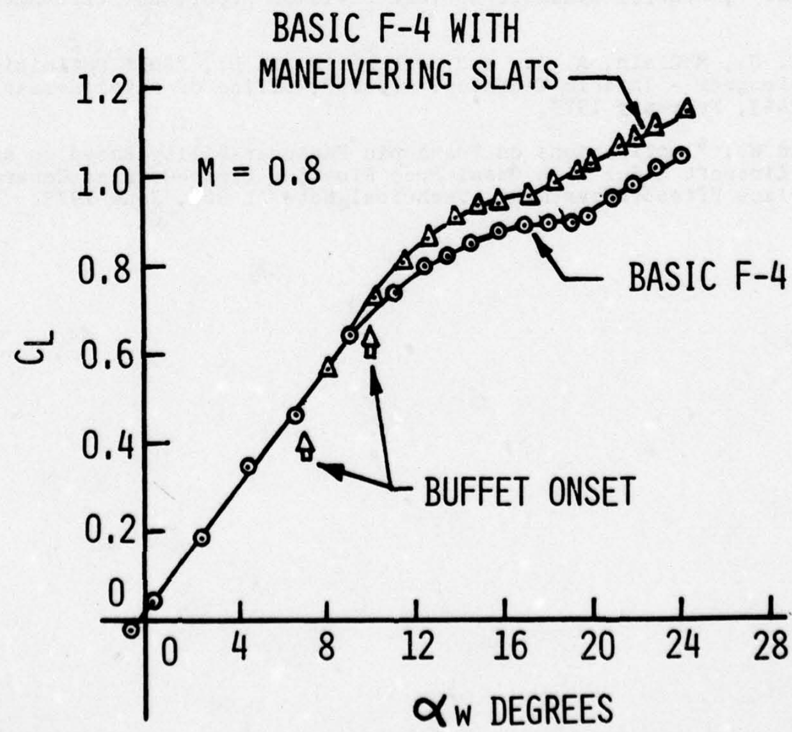

Fig.2 Effect of F-4E maneuver slats on buffet onset

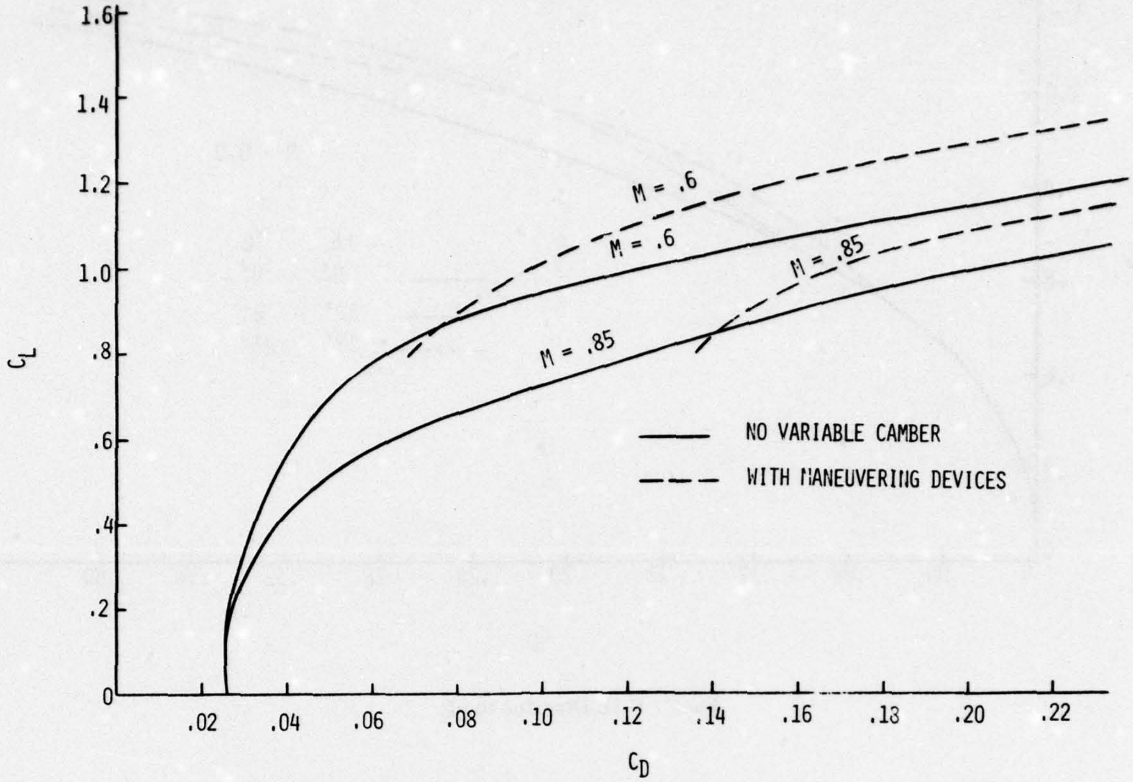

Fig.3 F-14A Drag polars

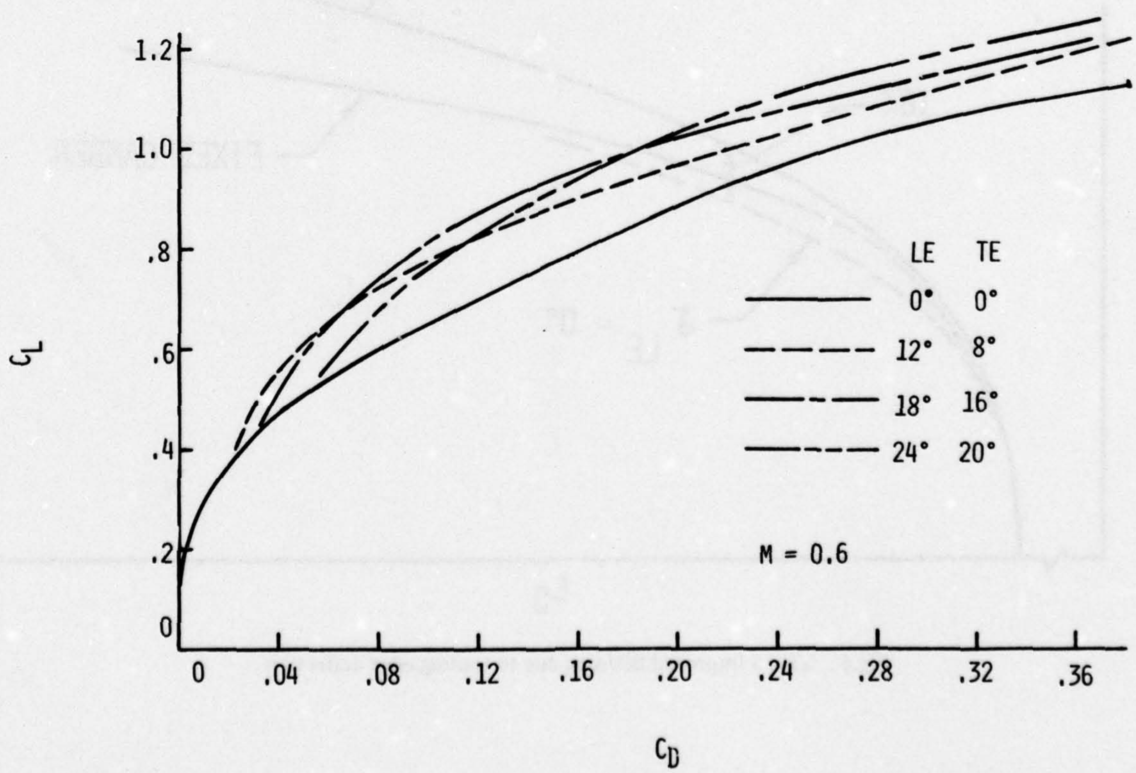

Fig.4 4 F-5E Drag due to lift

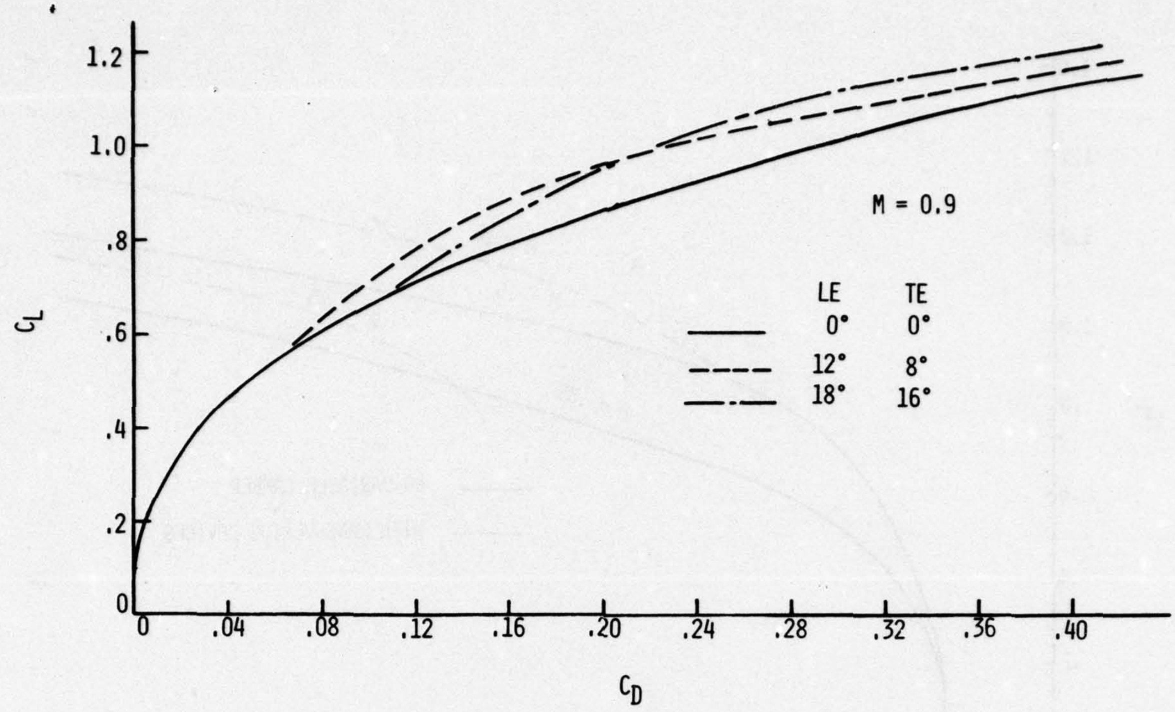

Fig.5 F-5E Drag due to lift

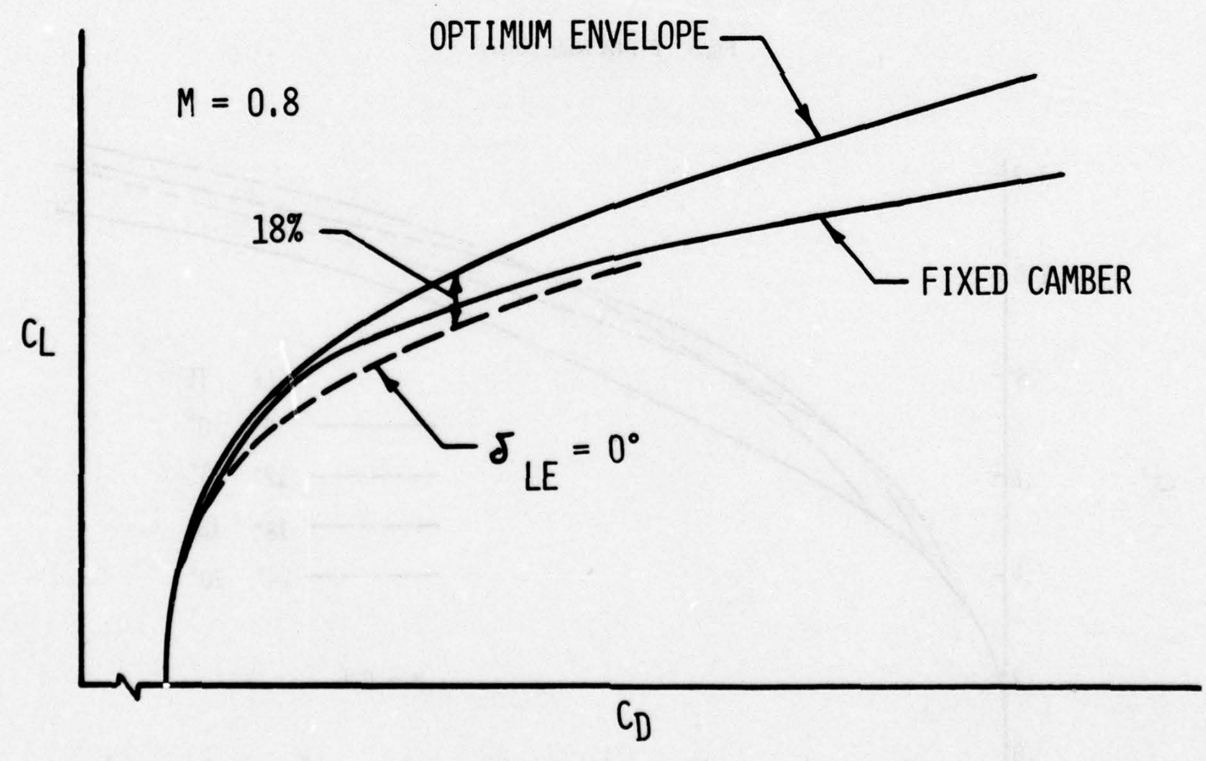

Fig.6 YF-16 Improved lift/drag due to leading edge deflection

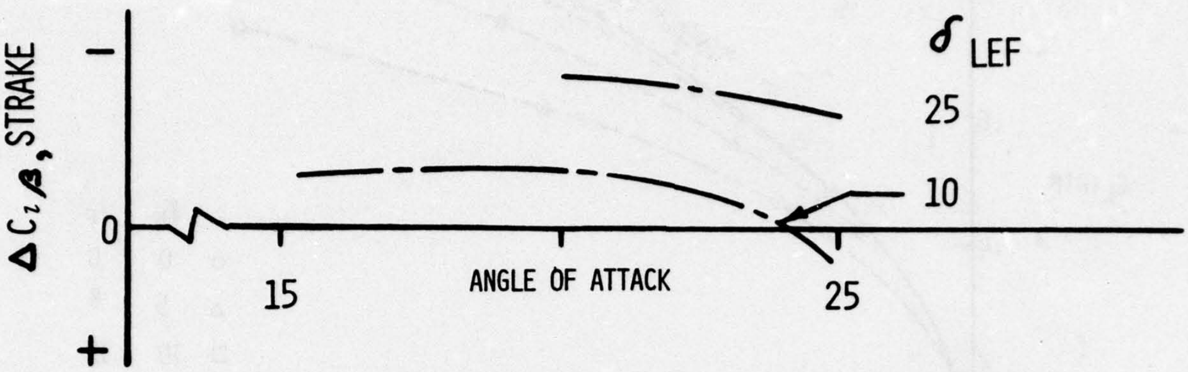

Fig.7 Strake contribution to lateral stability with leading edge flap deflection

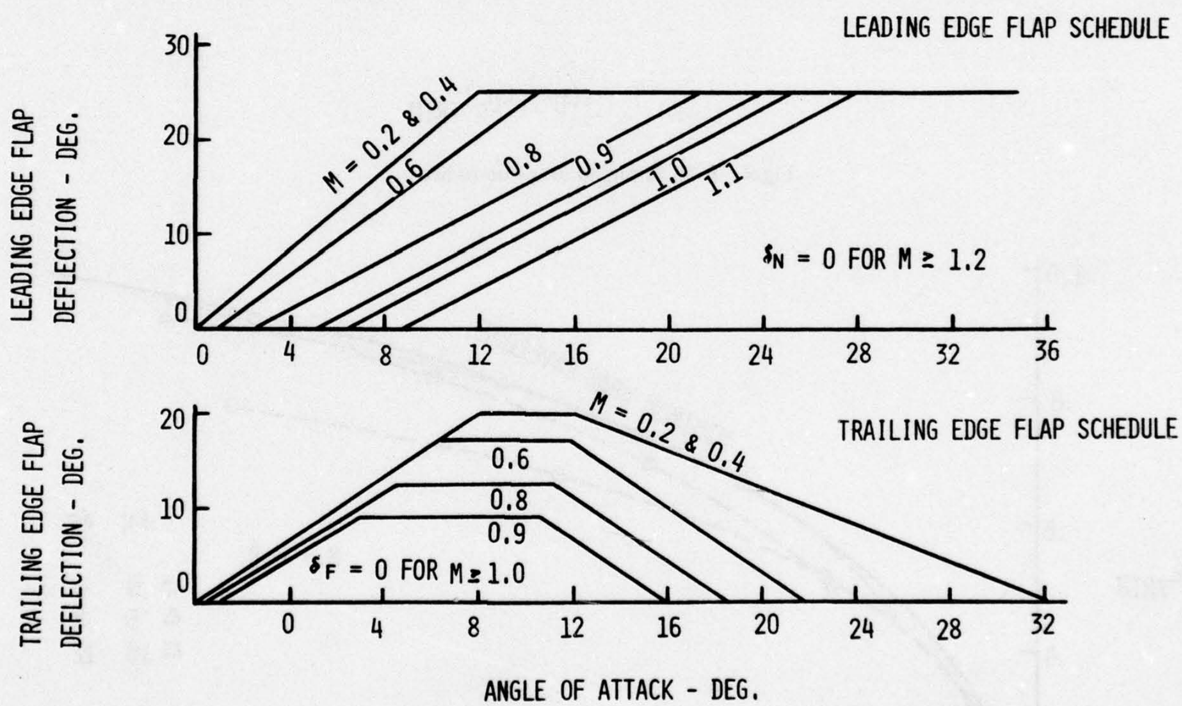

Fig.8 F-18 Flap schedules

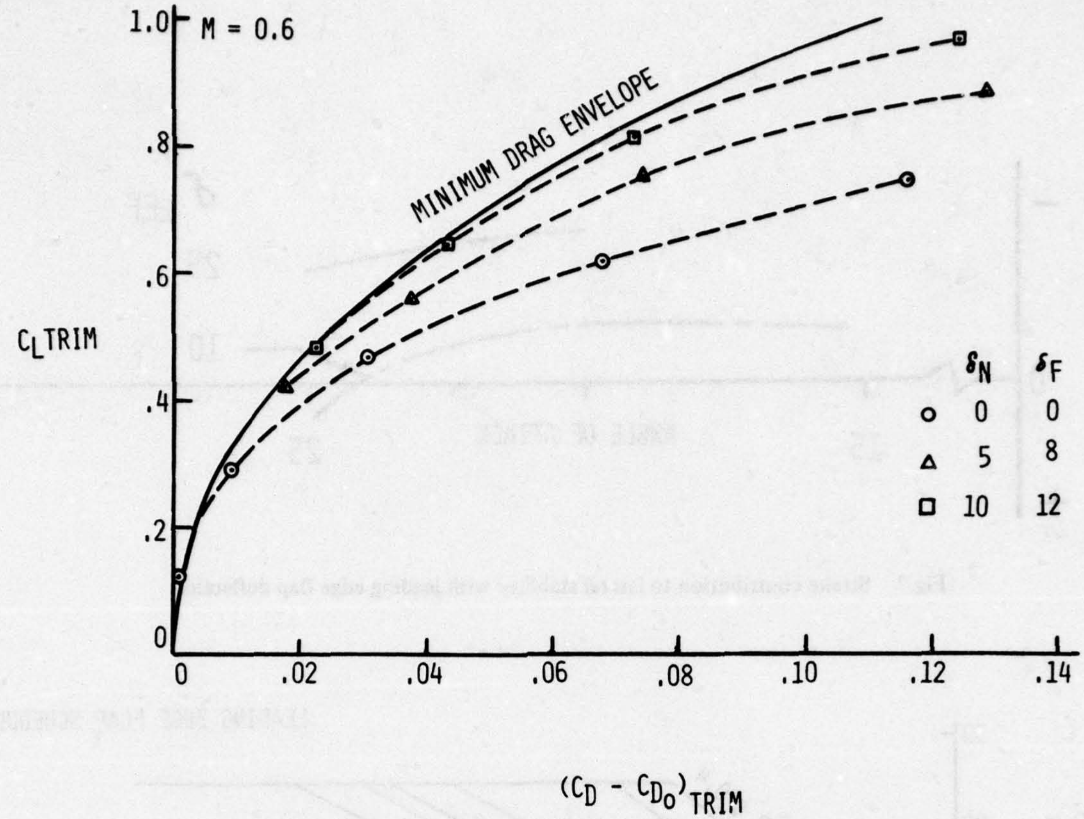

Fig.9 F-18 Trimmed drag-due-to-lift

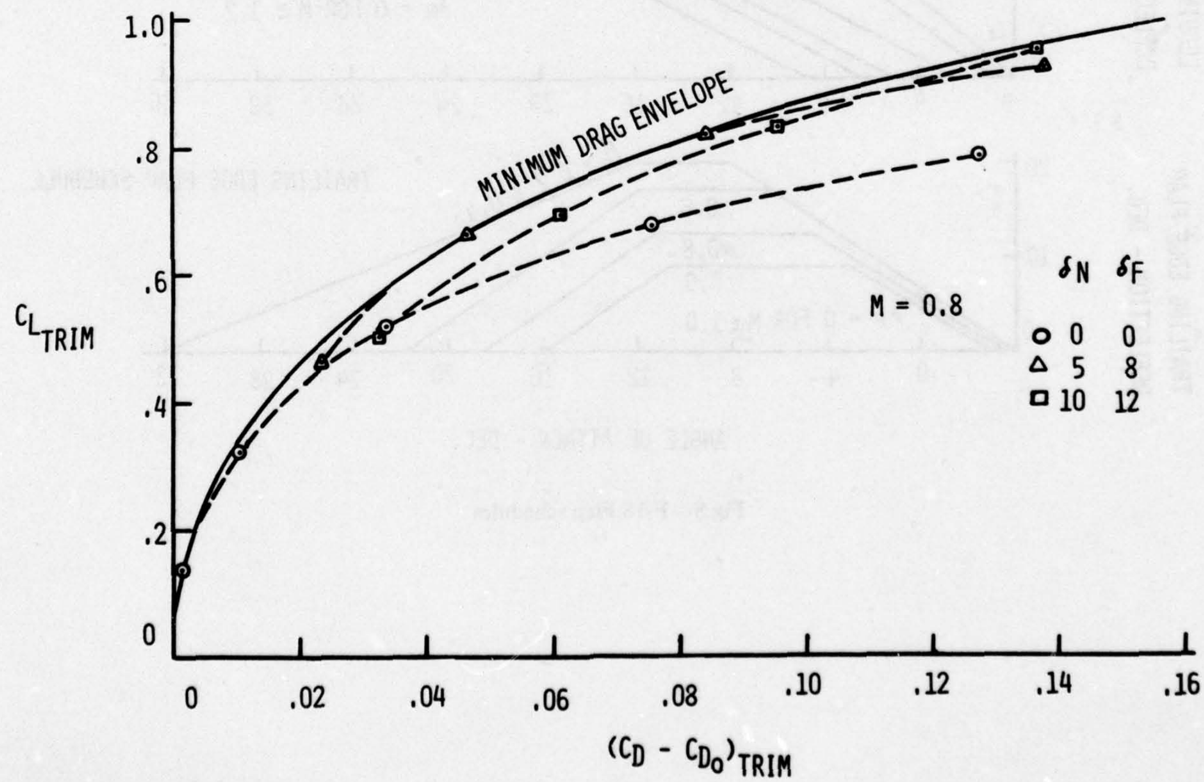

Fig.10 F-18 Trimmed drag-due-to-lift

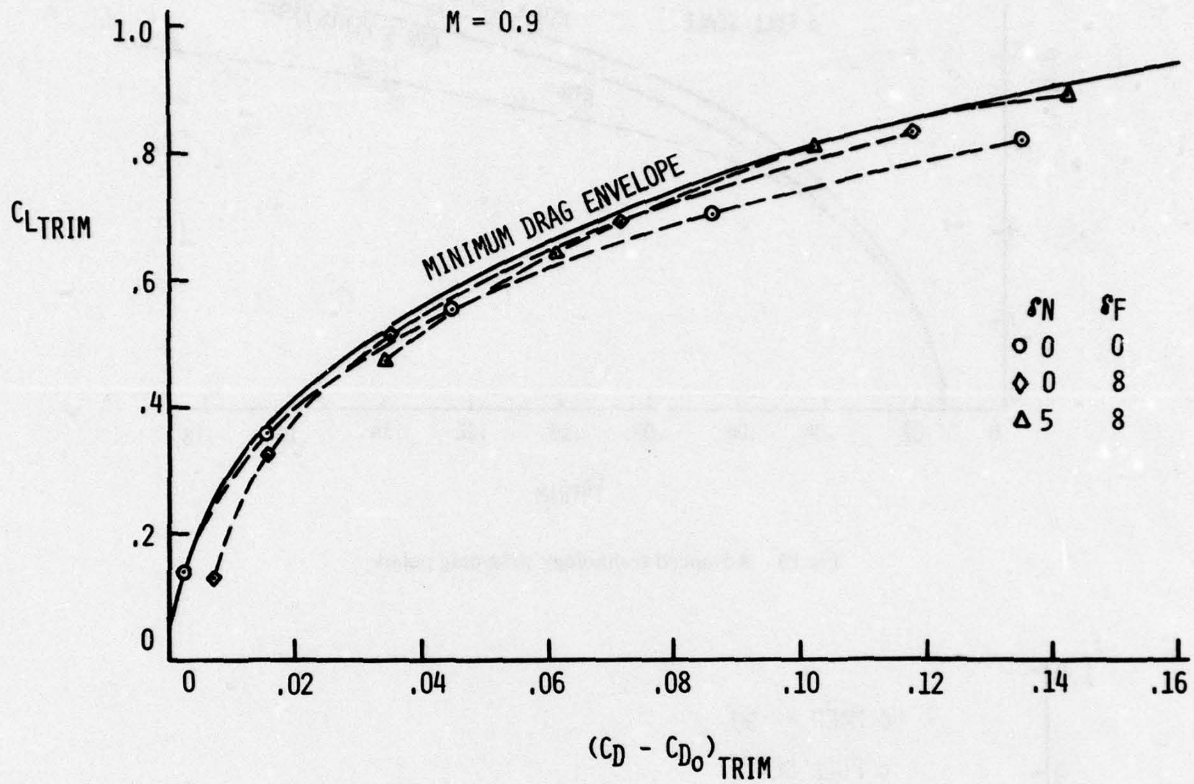

Fig.11 F-18 Trimmed drag due to lift, M = 0.9

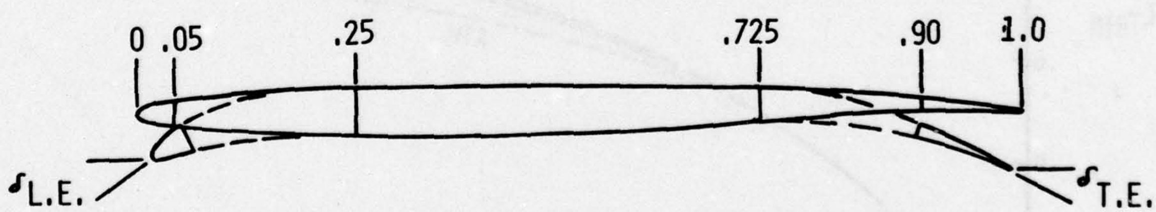

Fig.12 Advanced technology wing, airfoil section

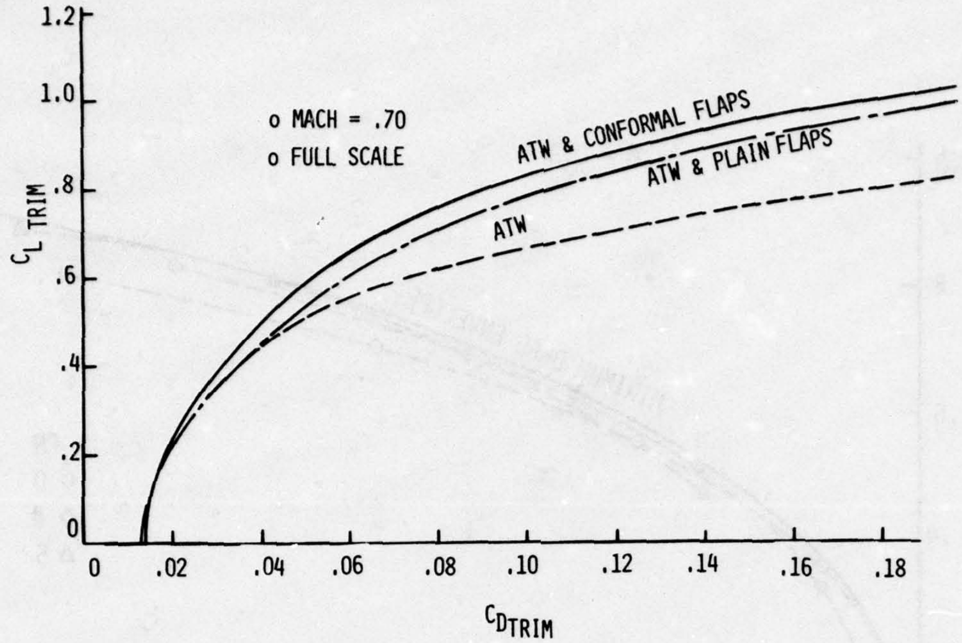

Fig.13 Advanced technology wing-drag polars

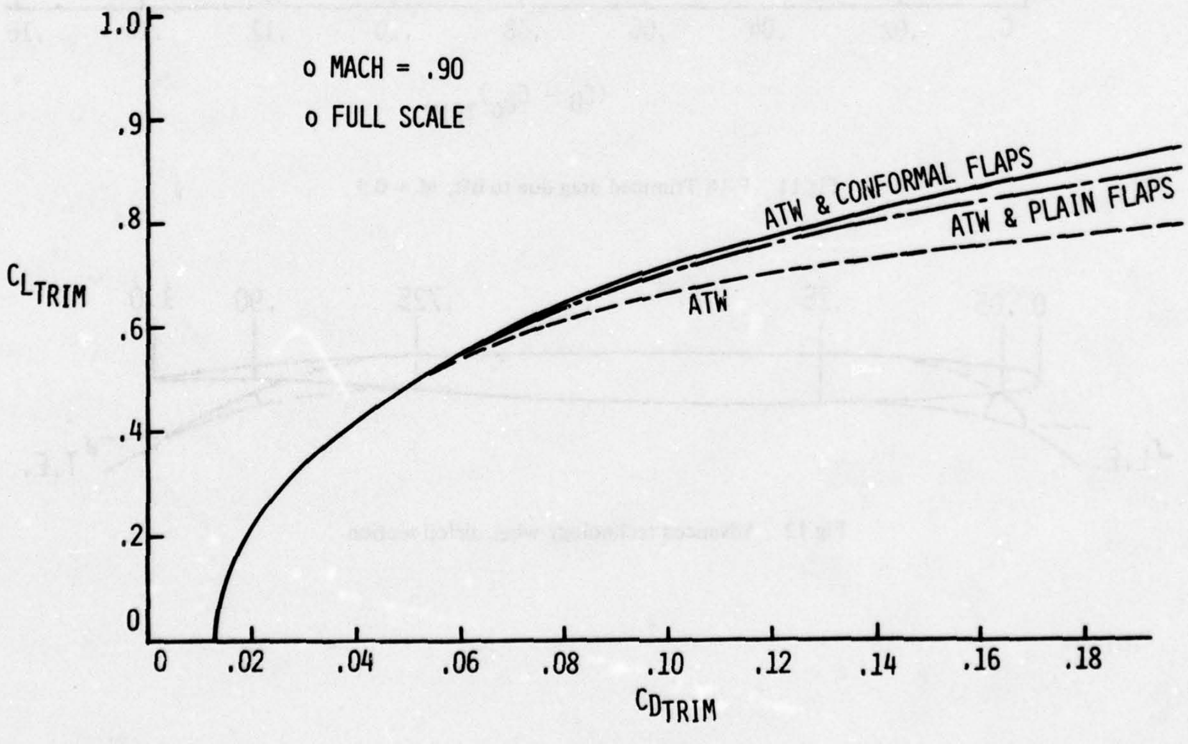

Fig.14 Advanced technology wing – drag polars

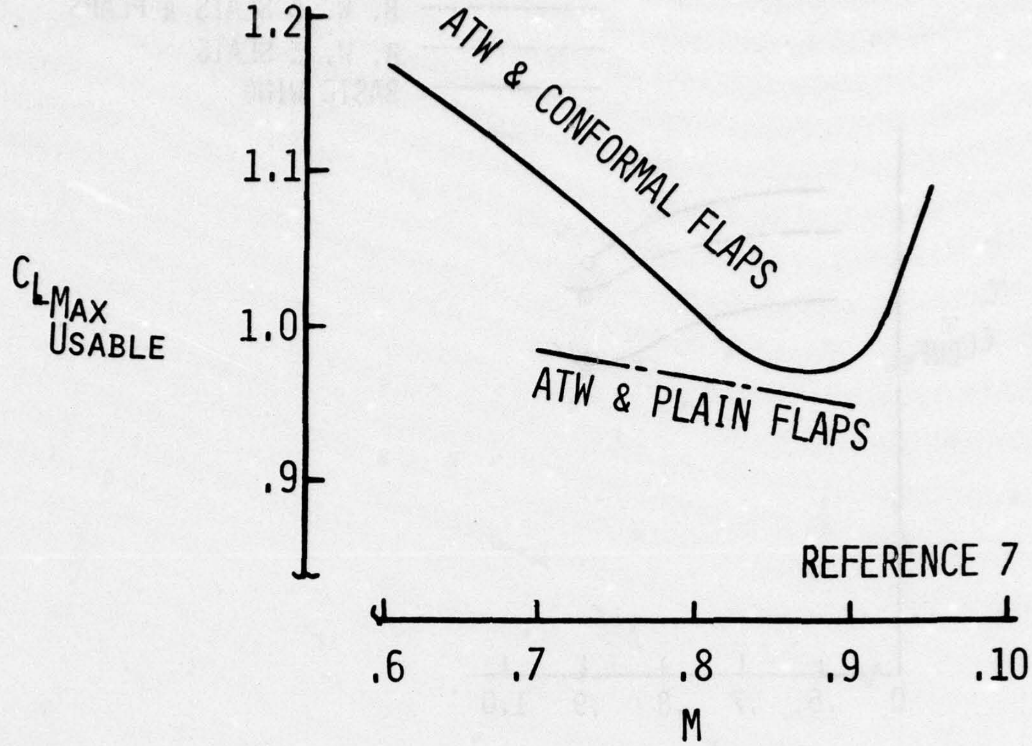

Fig.15 Advanced technology wing – maximum useable lift

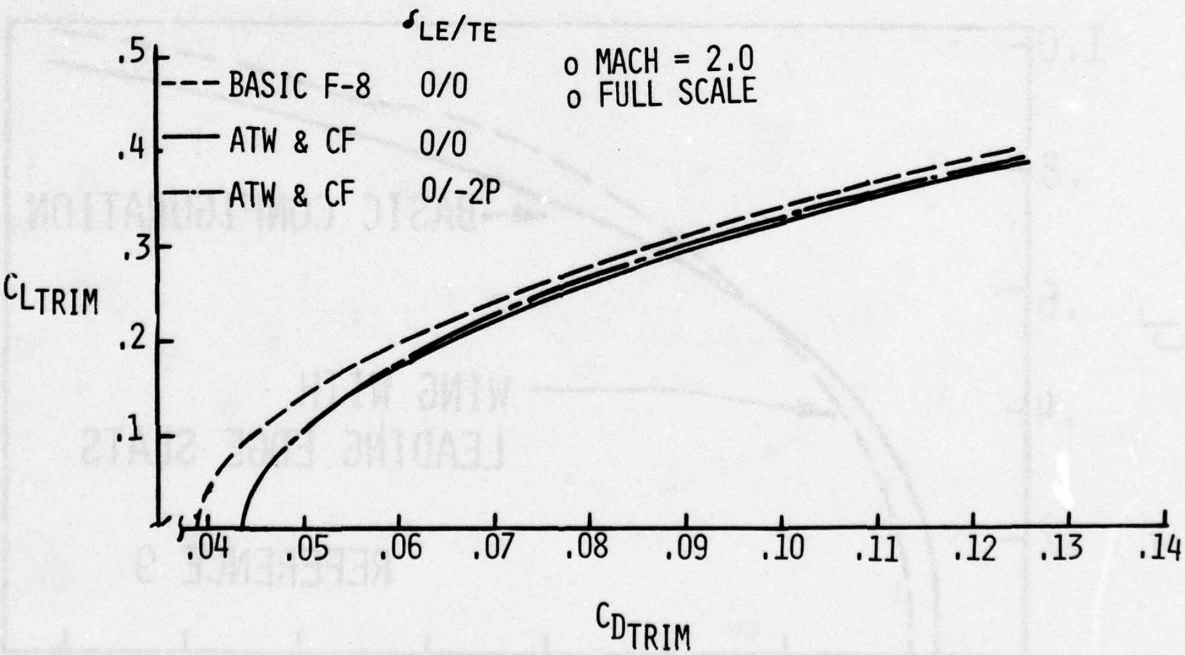

Fig.16 Advanced technology wing – drag polars

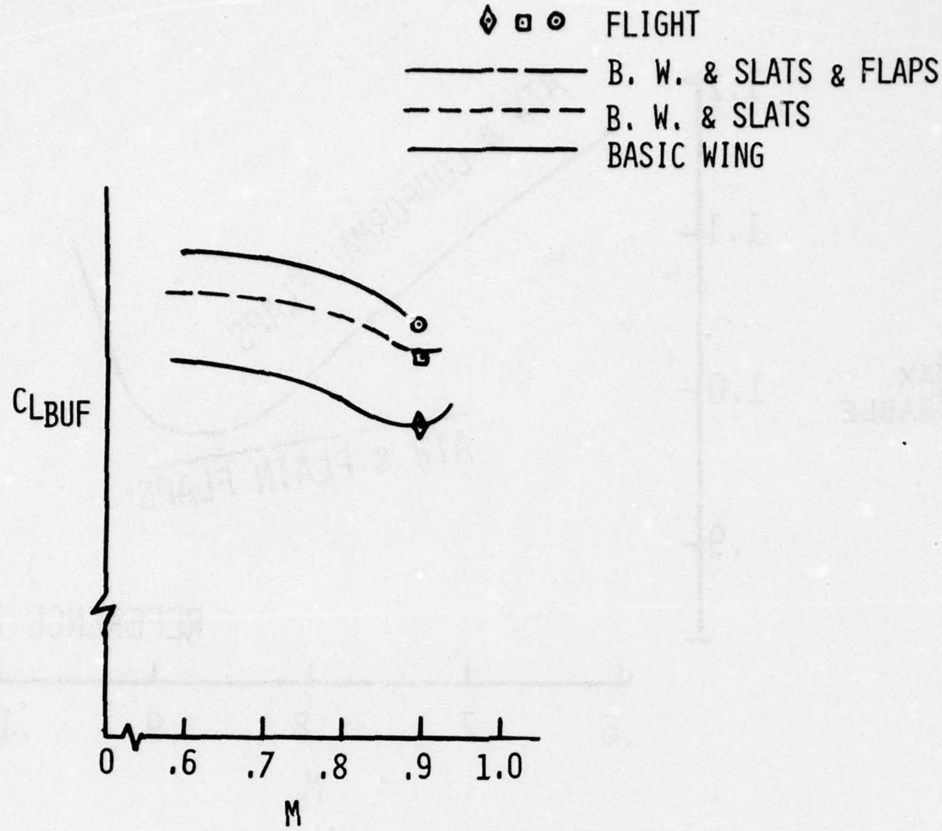

Fig.17 Slat and flap effects on buffet

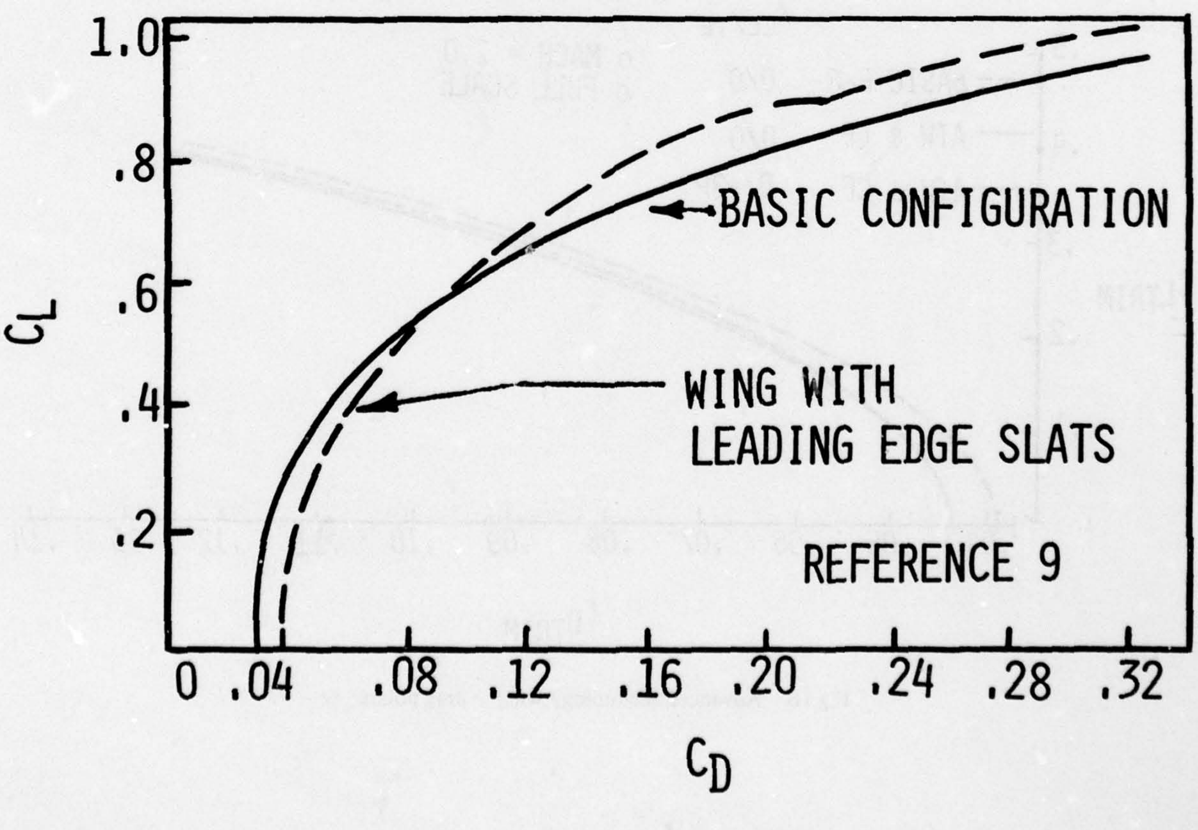

Fig.18 Influence of slats on drag polar

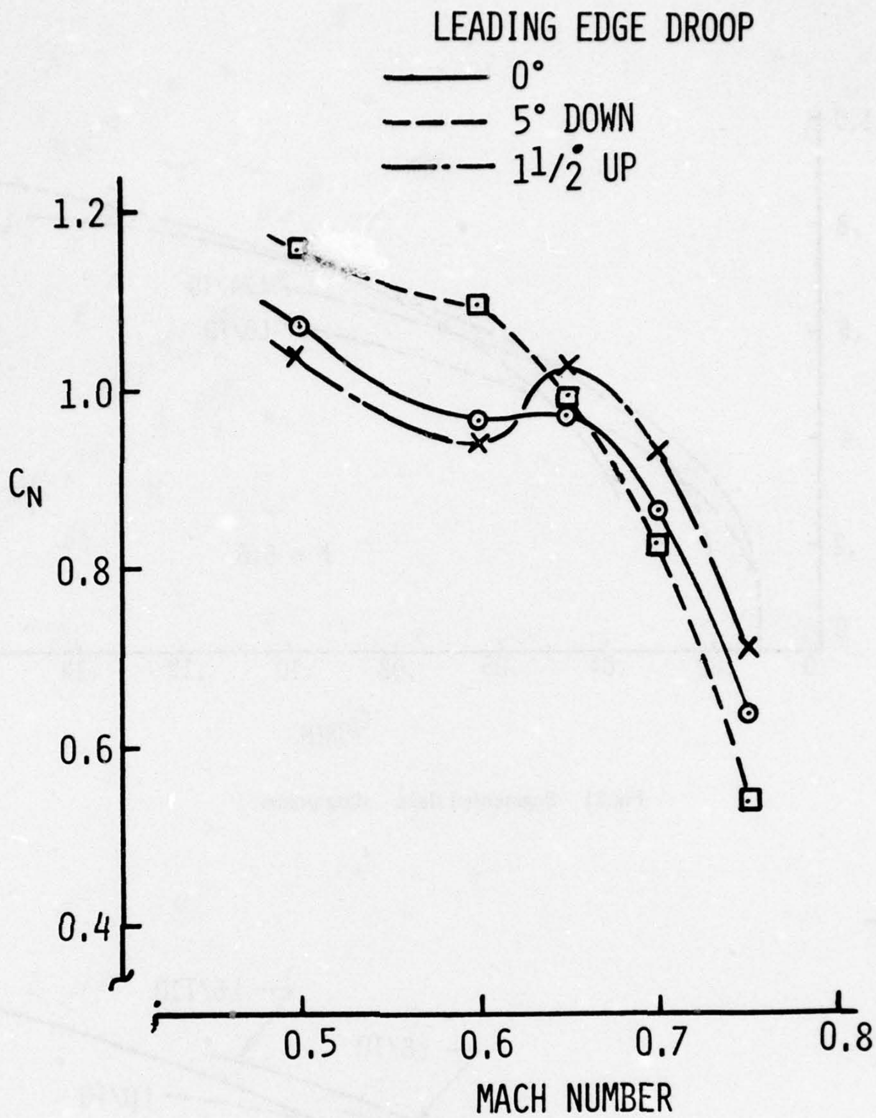

Fig.19 Raevam airfoil stall boundaries

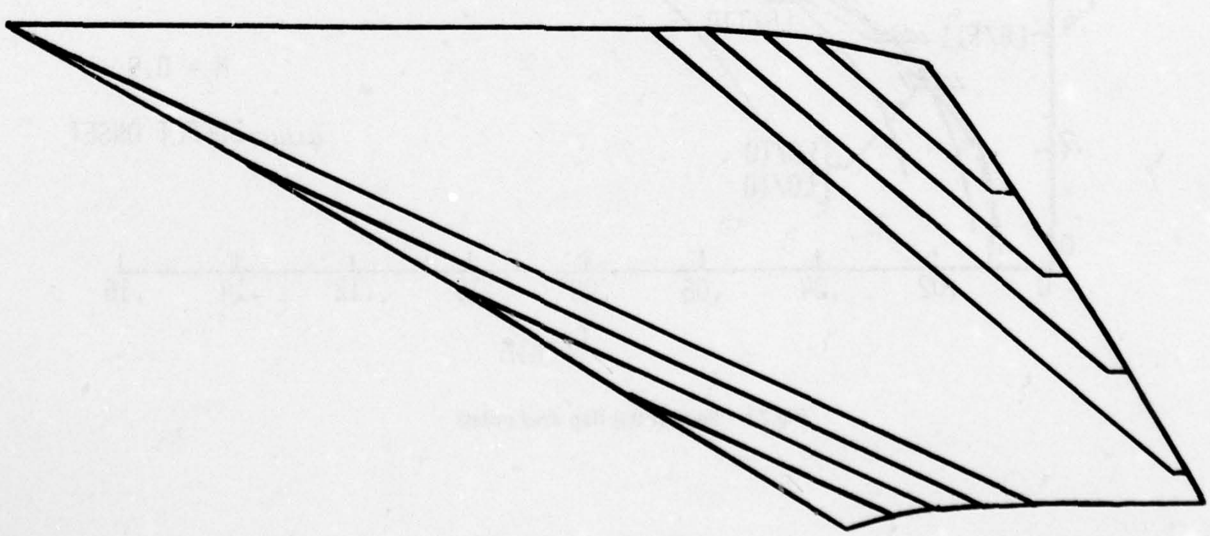

Fig.20 Segmented variable camber wing

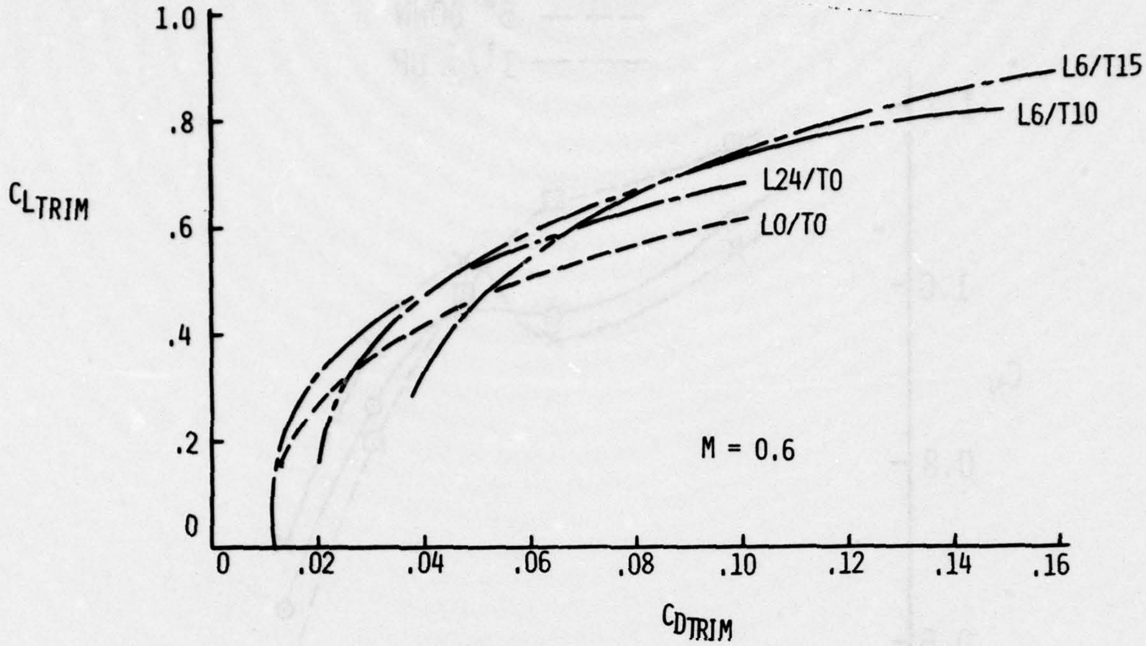

Fig.21 Segmented flaps – drag polars

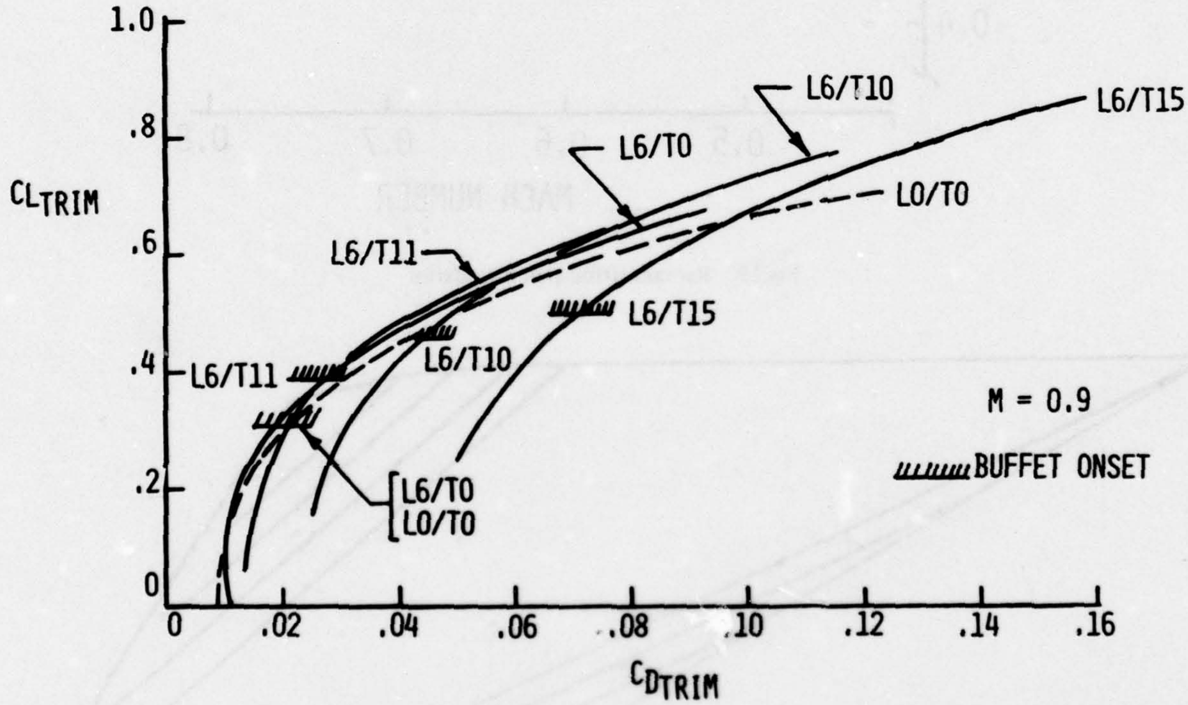

Fig.22 Segmented flap drag polars

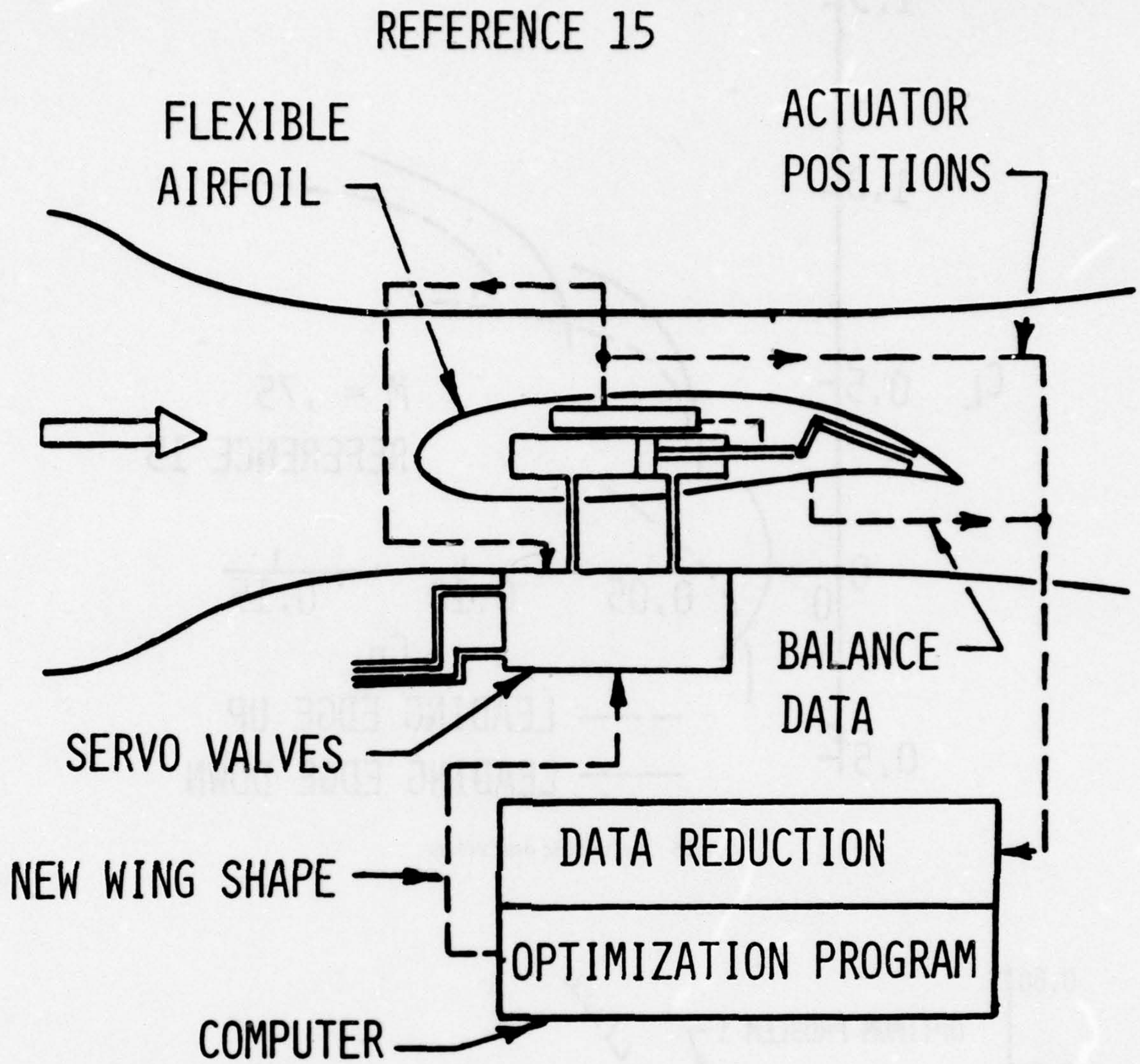

Fig.23 Self optimizing airfoil control system

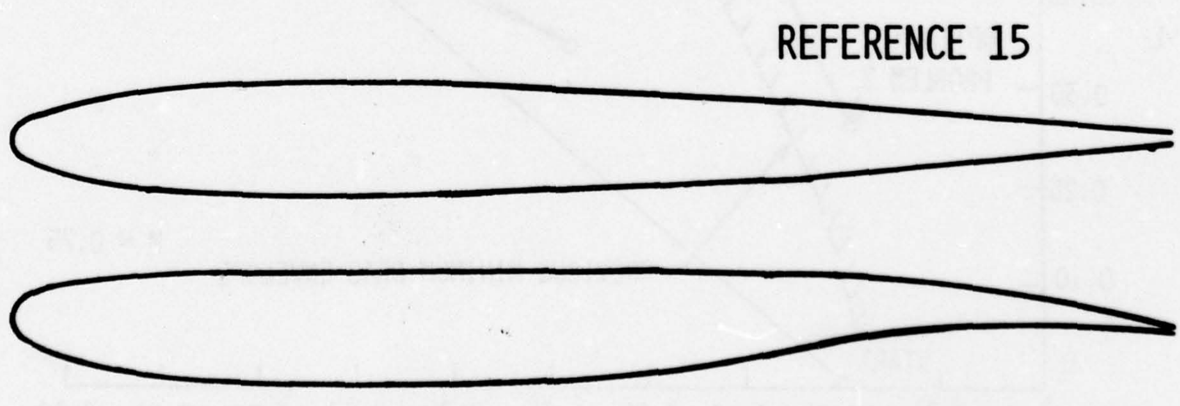

Fig.24 Self optimizing reference airfoils

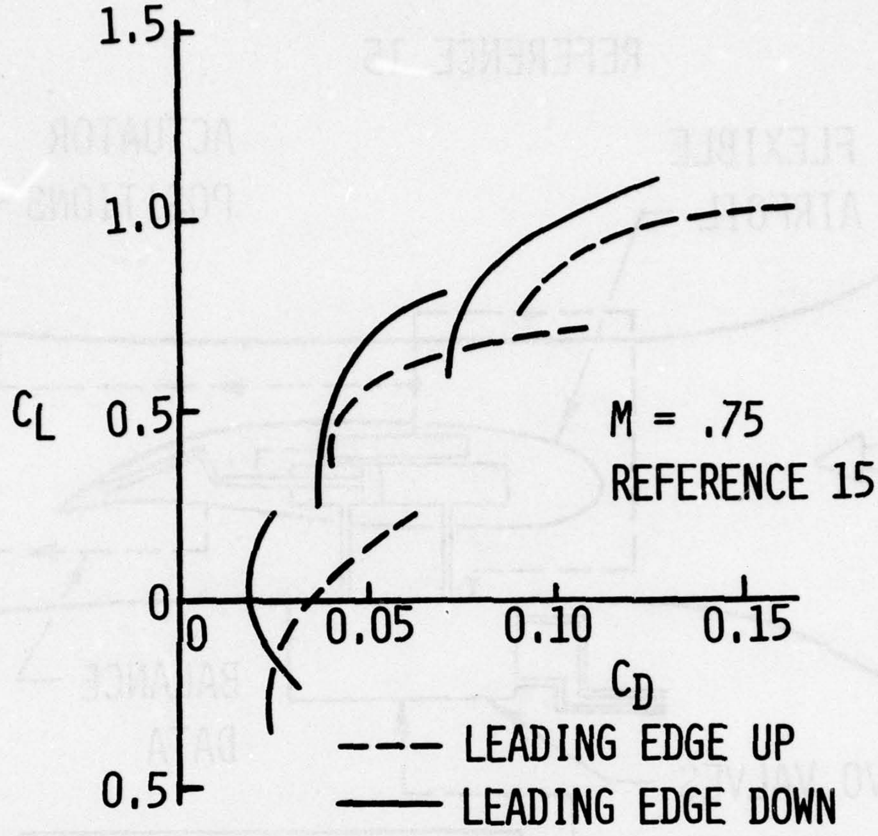

Fig.25 Parametric drag polars

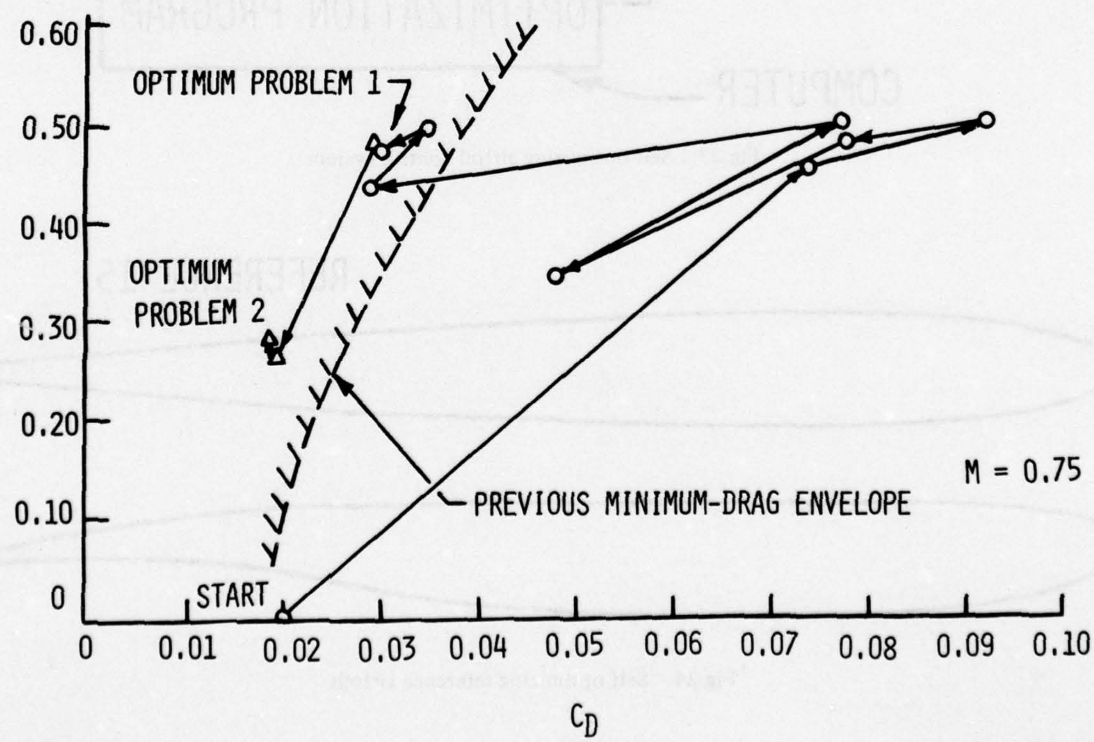

Fig.26 Self optimizing airfoil drag polar

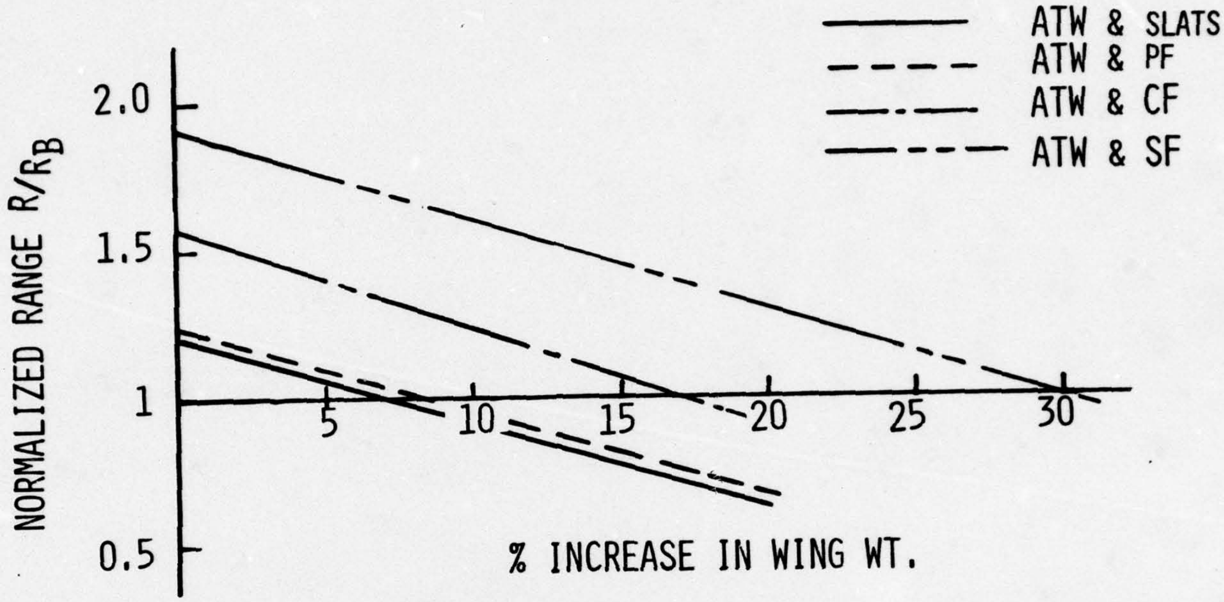

Fig.27 Range performance vs. wing weight increments

VARIABLE-CYCLE ENGINES FOR FIGHTER AIRCRAFT - ADVANCE IN PERFORMANCE AND DEVELOPMENT PROBLEMS*

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Summary:

In the introduction, the variety of requirements on engines for civil and military supersonic aircraft and the resulting interest in variable-cycle engines are explained. In addition, the variable-cycle engine concepts so far known and their function are described briefly.

For two concepts of variable-cycle engine for fighter aircraft, the data of which are available, the operating characteristics, performance data and leading design features are discussed. Further, these examples are used to contrast the advances offered by such engines against the problems to be overcome. It emerges that, whereas the aimed-at flexibility in operating characteristic and performance is feasible, the extra engine weight and complexity compared with conventional engines is very substantial.

For a fighter aircraft, a simplified comparison of effectivity is made between the installation of variable-cycle engines of the described concepts and two types of conventional engines with different bypass ratio. In line with findings published so far, it is shown that the extra weight to be expected with variable-cycle engines investigated so far, does not justify their future application. Even if further conceivable improvements in performance are achievable and if engine weight and complexity can be restricted simultaneously, variable-cycle engines will only have a real chance in mixed subsonic/supersonic missions of medium or long endurance.

Symbols

A	m^2	nozzle area
F	kN	thrust
M	kg/s	mass flow
R	km	mission range
W	kg	engine weight
SFC	$\frac{g/s}{kN}$	specific fuel consumption

Abbreviations

AB	afterburner
Dry	operation without reheat
RH	operation with reheat
MN	flight Mach number
SL	sea level
LP	low pressure
IP	intermediate pressure
HP	high pressure
LTJ	"leaky" turbojet
MTF	mixed-flow turbofan
VTJ	variable turbojet
VTF	variable turbofan

1. Introduction

Development of engines for combat/fighter aircraft, and for supersonic commercial aircraft as well, has not only led to enormous technical progress in the last two decades but also to considerable specialisation. Such engines have comparable thrust/weight ratios, turbine entry temperatures and overall pressure ratios, but the specialisation is characterized by a wide range of bypass ratios stretching from 0 to about 2.0. This results from the matching of the engines to various aircraft and missions, for which engine performance data and operating characteristics with due regard to installation effects serve as the criteria.

While requirements regarding the effectivity and economy of aircraft which operate both in the subsonic and in the supersonic range continue to become more stringent, the question of corresponding adaptability of the engines has been under discussion for some time. From the viewpoint of the mission planner and of the aircraft designer, there exist specific packages of requirements which in engines of conventional design can be met only in terms of a compromise. Thence, the big interest in engines with a variable thermodynamic cycle is well understood. The major requirements which are made, are summarized in Fig. 1 .

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2. Published engine concepts

A research programme [1, 2, 3]+ managed by NASA has been underway in the USA for some time to work out the fundamentals of economical and, simultaneously, low-ecological-impact engines for future supersonic commercial aircraft. A study/experimental programme [4, 5] on military engines with variable geometry is also being carried out on behalf of the US Air Force and Navy. A number of concepts have become known from these programmes. The concepts shown in Fig. 2 and 3 represent important basic types which show, at the same time, what fundamental possibilities exist for varying thermodynamic cycle parameters, as well as mass flow and thus performance data.

The "valved" concepts I - III shown in Fig. 2 are mainly envisaged for application in supersonic commercial aircraft.

In the case of Concept I, the two LP-compressors can be operated, by means of a changeover valve, in series or in parallel [3, 6, 8, 10].

- Operation in series results in a turbofan with moderate bypass ratio for supersonic cruise flight. In addition, duct burning of the bypass stream can be used in this mode to cater for the high thrust needed for takeoff, climbing and transsonic acceleration.
- Operation in parallel results in a three-stream turbofan with a high bypass ratio for subsonic cruise flight and landing.

A variable LP-turbine and internal variable flaps in the exhaust system are required to modulate the mass flows of first and second streams and, when necessary, of the third stream. Apart from modulating the engine mass flow, the convergent/divergent final nozzle permits optimum exploitation of thrust under all flight conditions. The latter applies also to all other concepts described.

In the case of Concept II, there is a changeover valve between the two LP-turbines [2, 6, 7, 9, 10, 11, 12]. This valve provides two modes of operation:

- Main stream and bypass stream are routed through the second LP-turbine, while the duct above the second LP-turbine remains closed. This provides a turbofan with high bypass ratio for subsonic cruise flight and landing.
- Bypass stream - possibly after duct burning - is routed through the second LP-turbine, while the main stream flows through the duct above the second LP-turbine. This provides two turbojets working in parallel for takeoff, climbing, transsonic acceleration and supersonic cruise flight.

Internal variable flaps in the exhaust system are required to modulate the mass flows of the first and second stream.

Concept III comprises a turbofan with a turbojet behind it [13]. In subsonic cruise flight and during landing the turbofan only is in operation, while the turbojet is shut down and the inner nozzle is closed. For takeoff, climbing, transsonic acceleration and supersonic cruise flight, the bypass stream of the turbofan is admitted to the turbojet, while the outer nozzle has to swallow the main stream of the turbofan. The turbojet functions as a thrust augmentor with particularly favourable fuel consumption.

There is one version of this concept without a changeover valve but with a secondary intake above the LP-compressor [14]. The flow to the turbojet passes through this secondary intake, there being no connection between turbofan and turbojet flow-wise.

Concepts I - III have two separate operating/performance ranges, depending on the two positions of the changeover valve, whereas all the concepts shown next, leaving aside the dry/reheat operation changeover, permit a continuous transition between the various configurations and operating ranges.

Concepts IV - VIII shown in Fig. 3 are envisaged for military and/or civil use.

In the case of Concept IV, the turbines and the internal flaps in the exhaust system can be given settings to provide a turbofan operating with two or three streams [9, 13, 14]. Operation on two streams *without* duct burning of the second stream is provided for supersonic cruise flight, *with* duct burning for takeoff, climbing and transsonic acceleration. Operation on three streams without duct burning is good for economical subsonic cruise flight.

This concept includes one version with three separate nozzles, [3, 14]. In another version, the inner and centre streams are mixed and led to an afterburner, while the outer stream remains separate [9, 11].

In the case of Concept V, the engine can operate as a turbojet or as a turbofan [12, 13, 14]. When low thrust is required, i.e., in subsonic cruise flight, the HP-system runs with high output, the entire engine working as a turbofan with duct burning shut off. When high thrust is required, the HP-system is throttled while duct burning is on at high temperature. Thus, the entire engine works as a turbojet. Apart from these extreme modes of operation, others are possible with appropriate harmonization of the burning temperature in both streams. The critical element of this concept is the variable second LP-turbine. By means of this component, not only the performance of the LP-compressor is modulated, but also the throttling of the entire system is influenced.

+ Numbers in square brackets refer to references at the end of the text

In the case of Concept VI, the settings of the turbines and of the internal flaps in the exhaust system can be changed to alter the bypass ratio within a limited range [7, 8, 9, 11]. A low-bypass-ratio setting is made for high thrust with or without duct burning. A higher bypass ratio is aimed at for dry part load, i.e., for subsonic cruise flight. The critical element of this concept is the variable HP-turbine, which must be considered particularly difficult to implement - at least at the present time.

A version of this concept with hot-stream reheat instead of duct burning is known [13], which seems more appropriate for military application.

Concept VII is also envisaged for operation as a turbojet at high dry or reheat thrust and as a turbofan at a dry part load, i.e., in subsonic cruise flight [14, 15]. The two streams expand through separate nozzles, so that reaction of the afterburner on the compressor section is, in principle, prevented. This concept will be the subject of further consideration.

A version of this concept with duct burning exists, mentioned in [12, 14].

Concept VIII has been known for a long time [16, 17, 18]. The variable turbine allows the specific dry thrust to be reduced at a large mass flow/high overall pressure ratio by lowering the turbine entry temperature, i.e., turbine mass flow capacity. In this way, SFCs at dry part load, i.e., in subsonic cruise flight, can be obtained which come close to those of turbofan engines with a low bypass ratio. With reheat, this concept is, in any case, an optimum one. This concept, too, will be covered in the later considerations.

3. Concept studies on variable-geometry engines

3.1 Requirements and development aims

The requirements and development aims given in Fig. 4 were evolved by fully utilizing the basic possibilities afforded by variable-geometry engines and taking into consideration practical experience with the latest turbofans in military service. Effects on engine performance and handling associated with engine installation are given special consideration, the more so as the impression exists that a good deal of the advantages aimed at are provided by the flexibility of the engine in combination with its integration in the airframe.

3.2 Engine concepts and basis of comparison

In line with the requirements and development aims described, two variable-geometry engines based on differing principles were designed and compared with two engines of conventional design with low and moderate bypass ratios. These four engines are shown schematically in Fig. 5.

In the case of the conventional engine with a "low" bypass ratio of 0.25 - called the "leaky turbojet" from now on - the bypass stream is restricted to the air requirement for afterburner cooling. This design provides advantages in terms of SFC with reheat. On the other hand, dry SFC is less favourable.

The conventional engine with a "moderate" bypass ratio of 1.3 - called the "mixed-flow turbofan" from now on - is designed for favourable dry SFC. Against this, its SFC with reheat is not as good as that of the leaky turbojet and aerodynamic stability with reheat is poorer.

The two variable-geometry engines were selected on the basis of the following considerations:

- The variable turbojet - shown in Fig. 3 as concept VIII - represents on the one hand the concept which is simplest in principle of those which have become known so far and provides optimum prerequisites at least with respect to handling - especially in connection with the afterburner. On the other hand, the variable HP-turbine involved may represent a development risk.

  As a refinement of concept VIII shown in Fig. 3, the afterburner is cooled by compressed air which is throttled to an extent preventing afterburner reaction on the compressor. The cooling air is shut down in dry operation.

- The variable turbofan - shown in Fig. 3 as concept VII - provides a considerable range of bypass ratios by keeping the two streams separated. The aerodynamic separation of the afterburner from the compressor section results in optimum stability with reheat. According to the aims given in Fig. 4, the engine works in reheat operation with a bypass ratio equal to zero, i.e., as a turbojet. The afterburner cooling air taken off downstream of the IP-compressor is throttled as in the case of the variable turbojet, thus preventing afterburner reaction on the IP- and HP-compressors. In this case, too, the afterburner cooling air is shut down in dry operation. In view of the considerable variation in the bypass ratio, a variable splitter between LP- and IP-compressors seems necessary. An important point is that the HP-turbine can have fixed geometry. Axial offsetting of the primary and secondary nozzles with flow around the primary nozzle prevents afterbody flow separation, and thus low afterbody drag can be expected.

In the case of the leaky turbojet and variable turbojet, a two-spool design was preferred, as the high pressure ratio of the LP-compressor results in a moderate pressure ratio of the HP-compressor. Admittedly, variable geometry is required in this case for both compressors to achieve aerodynamic stability. In the case of the mixed-flow turbofan and variable turbofan, the pressure ratio of the LP-compressor is relatively low, so that there is motivation for a three-spool design. Bleed can be used between IP- and HP-compressors of both mixed-flow and variable turbofans to increase the aerodynamic stability of the fixed-geometry compressors.

Fig. 6 gives the main engine design data, inasfar as they are significant for understanding the functioning of the four engines. For the purpose of a fair comparison, the same overall pressure ratio, turbine entry temperature and reheat temperature, as well as the same standard of component technology, were assumed for all engines. Admittedly, it is also assumed that the variable HP-turbine of the variable turbojet requires no lowering of its permissible turbine entry temperature vis-à-vis the other engines. Naturally, allowance is made for the losses in efficiency to be expected from variable components, inter alia. In the case of the leaky turbojet and mixed-flow turbofan the LP-compressor pressure ratio is prescribed by the mixing condition for both streams. In the case of the variable turbojet the LP-compressor pressure ratio results from the desired throttling of the afterburner cooling air, i.e., it is higher than in the case of the leaky turbojet. For the variable turbofan, there is a free choice of LP-compressor pressure ratio and a relatively low figure has proved attractive with a view to part-load SFC. In the case of the variable turbojet, the shut down of the afterburner cooling air in dry operation forces LP/HP-compressor matching for a higher overall pressure ratio than in reheat operation. In the case of the variable turbofan, the shut down of the afterburner cooling air in dry operation with appropriate IP/HP-compressor matching does not affect the overall pressure ratio, but results in an increase in bypass flow.

3.3 Performance data and operating behaviour

From the performance point of view, the characteristics of maximum thrust and of part-load SFC are of greatest interest. Fig. 7 shows the maximum thrust with and without reheat. It is known that for the same overall pressure ratio, turbine entry temperature and afterburner exit temperature as a function of flight Mach number and altitude

- the mixed-flow turbofan provides somewhat steeper thrust characteristics with reheat, i.e., higher maximum thrust in supersonic flight, than the leaky turbojet;
- the leaky turbojet, on the other hand, provides much higher dry maximum thrust.

The area between the leaky turbojet and mixed-flow turbofan characteristics shows the design latitude available in practice in the case of conventional engines.

As expected, in the case of the two variable-geometry engines reheat thrust characteristics similar to that of the leaky turbojet are obtained. While the throttling of the afterburner cooling air proved to be of rather negligible influence on performance, appropriate setting of the variable turbine geometry results in somewhat steeper thrust characteristics at high flight Mach numbers vis-à-vis the leaky turbojet.

The max. dry thrust of the variable turbojet is naturally somewhat higher than that for the leaky turbojet, as the bypass ratio is zero, once the afterburner cooling air has been shut down. As expected, the dry thrust of the variable turbofan is comparable with that of the leaky turbojet.

Fig. 8 shows two examples of the dry and reheat part-load characteristics of the four engines compared. The SFC characteristic of the variable turbofan shows clearly the transition from turbojet under maximum thrust conditions to turbofan under dry part load.

The variable turbojet has, under dry part load, a SFC of the same order as can be observed in the case of the leaky turbojet. In this context, it may be of interest that a variable turbojet design with a lower pressure ratio would result in roughly the same part-load SFC, whereas a much higher consumption would have to be accepted for maximum dry thrust. For this reason, the variable turbojet was designed for the same overall pressure ratio as the other three engines, unlike designs proposed in other publications [16, 17, 18].

After the information provided by the performance data for the non-installed engines, operating parameters which are important for installation now have to be discussed. Fig. 9 shows the mass flow characteristics over the entire dry and reheat thrust range. In the case of the two conventional engines, mass flow falls off steeply as dry part load comes down, resulting in increasing spillage drag owing to the intake capacity which has to be designed for maximum engine mass flow. This is an inevitable disadvantage in the case of conventional engines, owing to the clear connection between thrust and mass flow.

In the case of the two variable-geometry engines, on the other hand, the mass flow can be kept constant down to low part load, i.e., the same as maximum mass flow. Thus, spillage drag can be substantially reduced. In the range of very small thrusts, the setting characteristics of the variable components can be optimized to influence the engine mass flow in such a way that the lowest SFC is obtained with the installed engine. For a given thrust, a higher mass flow can be expected in this case than for optimization of the non-installed engine.

Fig. 10 is a plot of the nozzle throat area which, owing to the boattail angle associated with it, has a considerable effect on the afterbody drag of the aircraft. In the case of conventional engines, especially of the mixed-flow turbofan, considerable differences in the nozzle area with a correspondingly large boattail-angle range up to 20° for dry part load have to be mastered. In the case of the variable-geometry engines, on the other hand, the differences in nozzle area between maximum thrust with reheat and dry part load are relatively small, so that it is easier from the outset to optimize the fuselage afterbody with smaller boattail angles. Favourable conditions can thus be expected for afterbody aerodynamics, particularly in subsonic cruise flight.

Based on typical data on intake pressure loss, spillage drag and afterbody drag, Fig. 11 shows the SFC of the four engines compared at dry part load. The reheat range was omitted

in this case, as no significant relative shifts occur compared with the data for the uninstalled engines. In a thrust range which is important for subsonic cruise flight, about the same figure is obtainable with the variable turbofan as with the mixed-flow turbofan. Whereas high maximum dry thrust is obtainable with the variable turbojet, its SFC at dry part load comes close to that of the leaky turbojet.

In this comparison, the favourable influence which can be expected on the afterbody flow conditions in the case of the variable turbofan in view of its nozzle arrangement, as explained in Fig. 12 , is not included owing to a lack of information.

3.4 General arrangement and engine weight

As far as the variable turbofan is concerned, the performance data achieved could be considered a success, if a considerable excess weight and complexity were not involved at the same time. Fig. 13 shows the general arrangement of the conventional leaky turbojet and the mixed-flow turbofan compared with that of the variable turbofan. As far as possible, all three engines are designed on the basis of the same, currently available technology. The design of the variable turbofan, which can in no sense yet be considered an optimum one, served as a first attempt to become acquainted with the problems involved in designing such an engine. Conspicuous in the case of the variable turbofan are the much bigger radial dimensions of the core engine than in the case of the mixed-flow turbofan, and the greater length of the core engine than in the case of the leaky turbojet. In addition, the variable turbofan has a bigger spacing between LP- and IP-compressors with variable splitter, the extra casing to separate the bypass stream from the afterburner cooling air, as well as the complicated mechanism to change the settings of the primary and secondary nozzles. Despite the same overall pressure ratio as in the case of the conventional leaky turbojet and mixed-flow turbofan, the variable turbofan also requires a bigger design effort on the core-engine turbomachinery as, in this case, owing to the bringing-in or shutting-down of the afterburner cooling air stream, the three compressors do not reach their highest pressure ratios simultaneously for maximum dry thrust or maximum thrust with reheat.

Including everything which the four extra control variables involve, the variable turbofan is some 30 % heavier than the leaky turbojet and mixed-flow turbofan, which are of roughly equal weight. Apart from the complexity of the variable turbofan, this weight is naturally not acceptable.

Although this result is disappointing at first sight, new thinking on the matter shows that there is still considerable latitude for improvements. It is expected that not only an improvement in the performance data - particularly at dry part load - can be obtained through appropriate redesign of the variable turbofan but that the weight penalty vis-à-vis the leaky turbojet and mixed-flow turbofan can presumably also be brought down to 10 - 15 %.

In the case of the variable turbojet not shown here, only a slight weight penalty of 5 - 7 % is expected vis-à-vis the conventional leaky turbojet. This results from the LP-compressor having a higher pressure ratio and the turbines having to be designed for variable geometry.

4. Engine evaluation

As shown in Fig. 14 , a rough comparative evaluation of the engines under consideration was made on the basis of an imaginary multi-role combat aircraft for the missions "Interception", "Air Superiority" and "Battlefield Interdiction/Close Air Support". For the four alternative engines compared, the weight of engine + fuel and thus the takeoff gross weight were taken to be the same, each mission being a special case. As a basis for comparison, which is in principle arbitrary, it was also assumed that the called-for mission ranges can be achieved by the mixed-flow turbofan.

Due to the different thrust characteristics of the alternative engines, their design thrusts differ to some extent. In order to emphasize the important role of the excess weight of both variable-cycle engines, their thrust/weight ratios were introduced parametrically. The leaky turbojet and mixed-flow turbofan were assumed to be specifically equally heavy, in line with the current state of the art.

Fig. 14 shows the ranges, as defined differently for each of the three missions, achievable with the alternative engines. If the variable turbofan could be built to have the same specific weight as the conventional engines, as well as the variable turbojet, it would be at least as good as or superior to them, depending on the mission. However, the equality or superiortiy of the variable turbofan in all missions is lost, as soon as it has a slight extra weight of 10 %. For all three missions considered, the variable turbojet does not promise any progress in aircraft performance. This holds true, even if the weight of this engine were the same as that of the conventional leaky turbojet.

Naturally, Fig. 14 covers only part of the evaluation which has to be carried out, especially as important criteria such as "turn rate" and "specific excessive power" must also be included. Furthermore, aspects relating to handling qualities of the engines should not be underestimated. In this context, it should also be emphasized that, at least in the case of the variable turbofan, i.e., going beyond the possibilities of the conventional engines, specific control measures to improve the compatibility of the compressor section with inlet distortions are possible. Moreover, with regard to extreme power and bleed off-take from the HP-system, for instance in an emergency case with one engine failed, maximum power and bleed off-take with minimum effects on the engine performance can be obtained by setting the configuration of the remaining engine for max. dry thrust. Finally, shorter acceleration times can be expected with variable-geometry engines than in the case of conventional engines [19].

5. Performance potential

Whereas the performance and weight data for the variable turbojet and variable turbofan shown so far reveal the position reached at present in the studies, they should not be considered as a definitive or fundamental pronouncement on the potential inherent in variable-geometry engines. At least in the case of the variable turbofan, it can be expected that the SFC at dry part load, i.e., for the important case of subsonic cruise flight, can still be improved. In Fig. 15 the hypothetical improvements on the variable turbofan in the SFC and mass flow characteristics are shown. Admittedly, there can be no doubt that an air mass flow which is kept constant at a high overall pressure ratio and high turbine entry temperature right down to small thrust as a prerequisite, implies a progressive increase in the bypass ratio. Whereas this parameter combination is thermodynamically possible, it is nevertheless scarcely obtainable with actual turbomachinery - all the more as the engine weight will be another restricting factor. It is therefore anticipated that it will only be possible to shift the SFC characteristic of the variable turbofan somewhat in the direction of the drawn-in characteristic of the hypothetical variable turbofan.

Besides this, it is basically possible to dispense with the afterburner in the case of the variable turbofan and thus to obtain very good SFCs at high thrust. The hypothetical SFC/thrust characteristic of this dry variable turbofan has a shape analogous to that of the variable turbofan with afterburner in dry operation. It can be seen that this concept, which is fascinating at first sight from the angle of design simplicity and handling qualities of the engine, has two serious drawbacks:

- On the assumption that the SFC of the variable turbofan cannot be improved much at low dry part load, rather higher SFCs will have to be accepted in subsonic cruise flight with the dry variable turbofan.
- Owing to the substantially higher air mass flow, the dry variable turbofan will certainly be much heavier than the version with afterburner. Furthermore, in view of the bigger air intake, substantial negative effects must certainly be expected on the design of the whole aircraft.

Résumé:

In view of the disappointing results so far of the studies on variable-cycle engines, the question must be asked in what direction further efforts should be guided.

Fig. 16 is a highly simplified summary of data available so far on SFCs in subsonic cruise flight and weights of variable-cycle engines. A comparison on this basis is sensible, particularly as, for all concepts included here, roughly the same behaviour in reheat operation, namely that of a turbojet with corresponding optimum SFC, is obtained. Conspicuous is the much too high weight of all variable-cycle engines vis-à-vis the conventional leaky turbojet taken as a basis of comparison. However, the comparison also shows there is no chance of crossing the break-even line through further improvement of the dry SFC. Future efforts must rather be directed, in the first place, to reducing engine weight.

Admittedly, a fair assessment of the variable-cycle engines requires allowance to be made for some other criteria already touched upon concerning, in particular, installation effects and handling qualities.

Summing up, the state of knowledge achieved on variable-cycle engines can be set down roughly as shown in Fig. 17 .

Acknowledgements

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Application In Commercial Supersonic Aircraft

- favourable SFC in supersonic and subsonic cruise
- high thrust in transsonic acceleration
- optimum engine integration, i.e.
  adaption of intake capacity and engine mass flow
  favourable afterbody/nozzle aerodynamics
- acceptable noise on take-off and landing
- low pollutant emission

Application in Combat/Fighter Aircraft

- favourable SFC in supersonic/subsonic flight, with/without reheat
- favourable thrust characteristics
- optimum engine integration (see above)
- optimum handling qualities

Fig. 1 Variable-Cycle Engines, Motivation and Development Aims

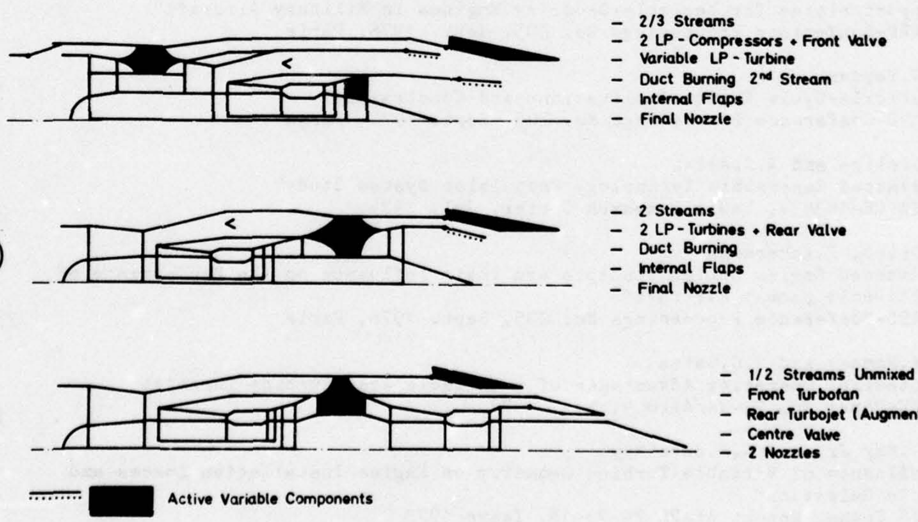

Fig. 2 Basic Types of Variables-Cycle Engines for Commercial (SST) Application (Schematic, Simplified)

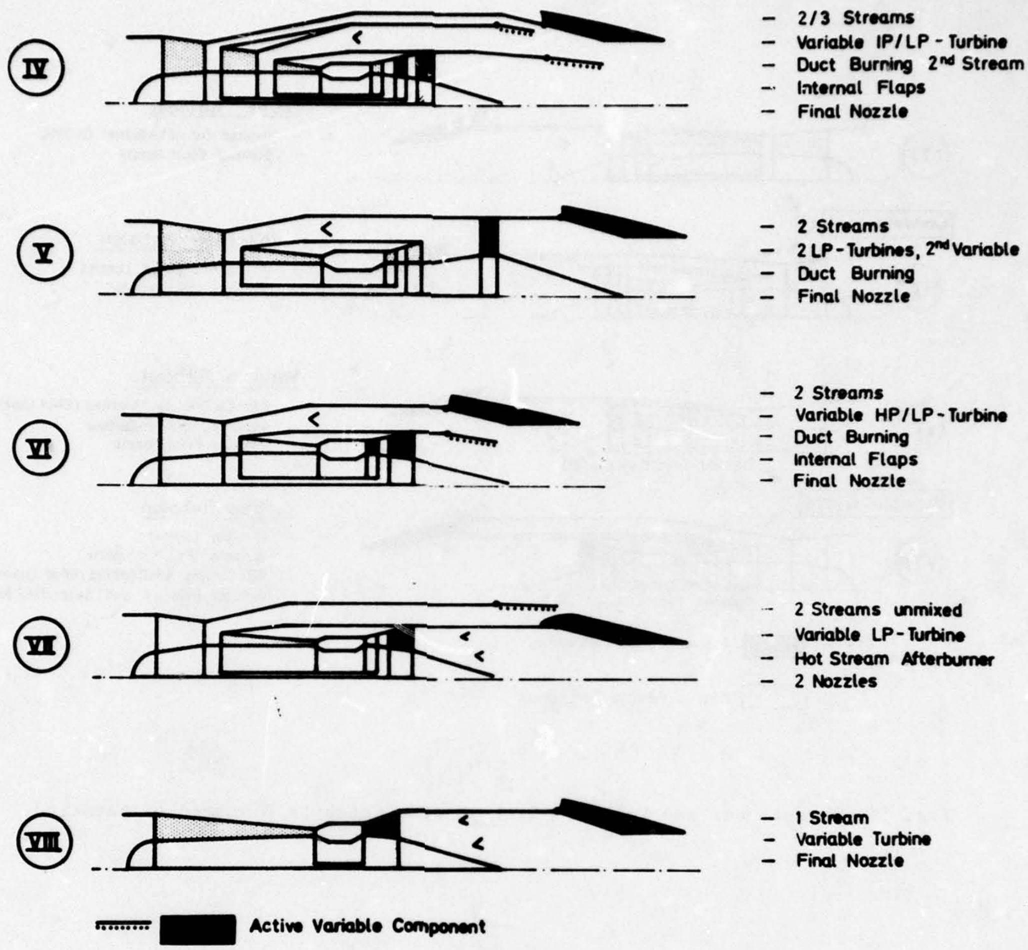

Fig. 3 Basic Types of Variable-Cycle Engines for SST or Military Application (Schematic, Simplified)

Performance
- optimum SFC with reheat (like turbojet)
- optimum SFC without reheat (like turbofan)
- high max. dry thrust, related to max. reheated thrust (like turbojet)

Envelope and Handling Qualities
- maximum altitude with reheat (like turbojet)
- high aerodynamic stability on afterburner light up

Installation
- Optimum matching of intake capacity and engine mass flow
- good compatibility with intake distortions
- favourable afterbody/nozzle aerodynamics
- low effect of bleed/power off-take on engine performance and stability

Engine Design
- acceptable weight and cost
- reasonable complexity including control unit

Fig. 4 Variable-Cycle Engines for Combat/Fighter Aircraft, Requirement and Development Aims

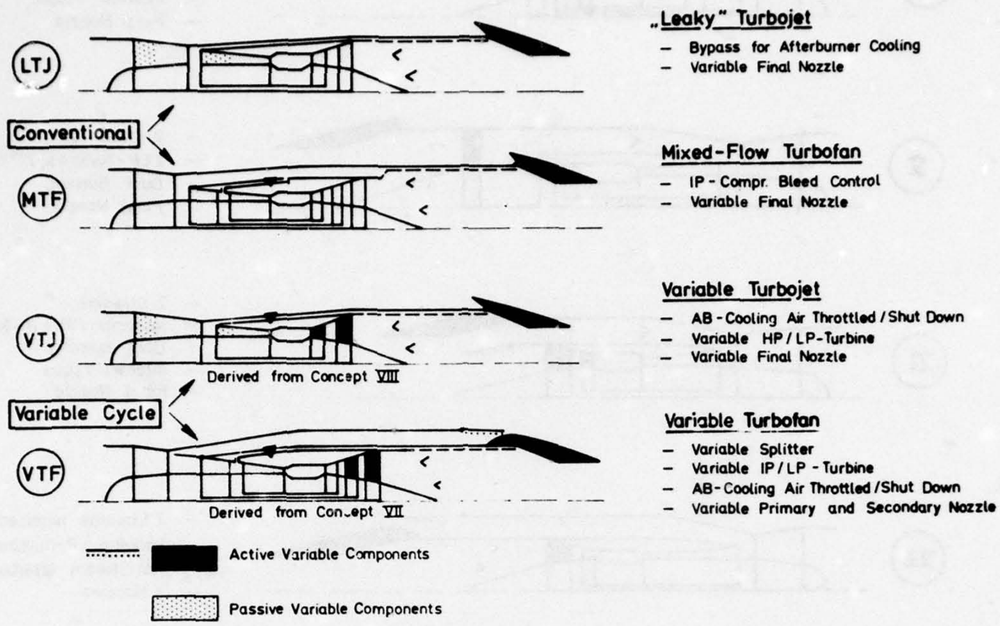

Fig. 5 Conventional and Variable-Cycle Engine Concepts Compared (Schematic)

SL/MN = 0,8/ISA/Uninstalled		Conventional				Variable Cycle			
Max. RH (Max. Dry)		LTJ		MTF		VTJ		VTF	
Thrust	kN	80	(42,7)	80	(31,8)	80	(50,0)	80	(42,8)
Mass Flow	kg/s	81		94		82		82	
Turbine Entry Temp.	K			1600	(1565)				
Reheat Temperature	K			2050	-				
Overall Pressure Ratio		20	(19,1)	20	(18,9)	16	(20)	20	(18,9)
Bypass Ratio *		0,25	(0,25)	1,3	(1,3)	-		0	(0,29)
Afterburner Cooling Air **		included in second stream				0,25 **	(o)	0,25 **	(0)
Pressure Ratios :									
LP-Compressor		3,3		2,4		4,3	(4,5)	1,9	
(IP+) HP-Compressor		6,1		8,3		3,7	(4,4)	10,5	
Nozzle/Engine Entry		2,9		2,1		3,0	(3,9)	3,0	(3,4) hot stream
									(1,9) cold stream

* Related to HP-Compressor

** Throttled in Order to Avoid Afterburner Reaction or Compressor Section

Fig. 6 Main Design Parameters of Engines Compared

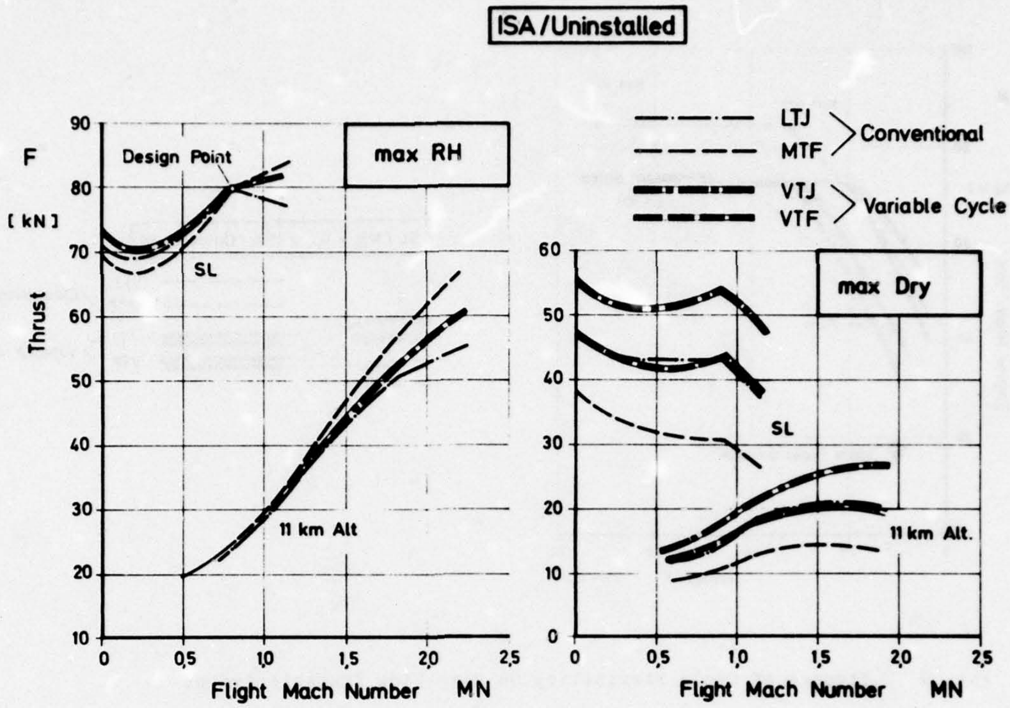

Fig. 7 Comparison of max. Thrusts with and without Reheat

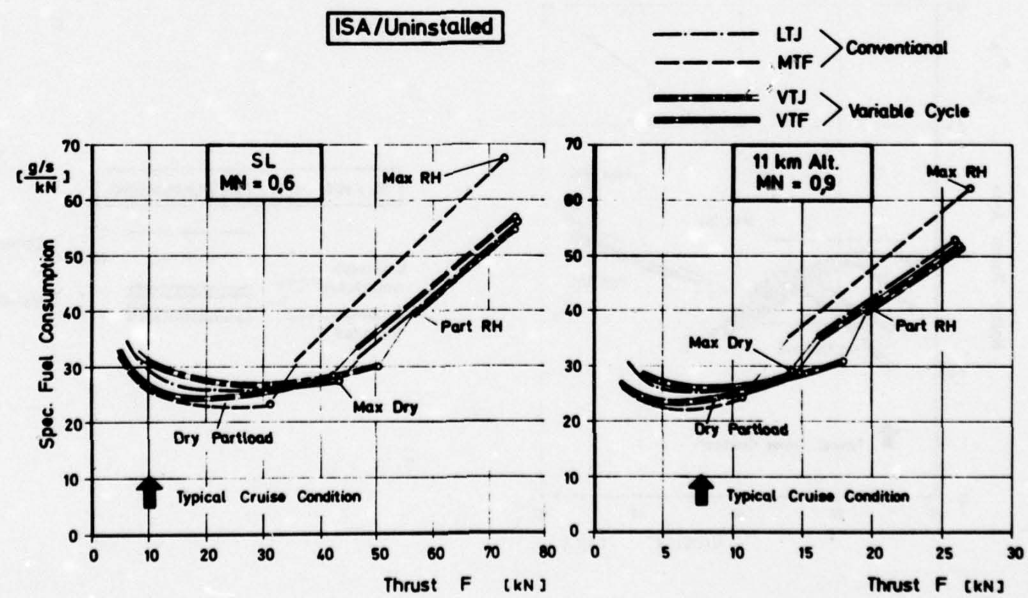

Fig. 8 Comparison of Partload SFC-Characteristics

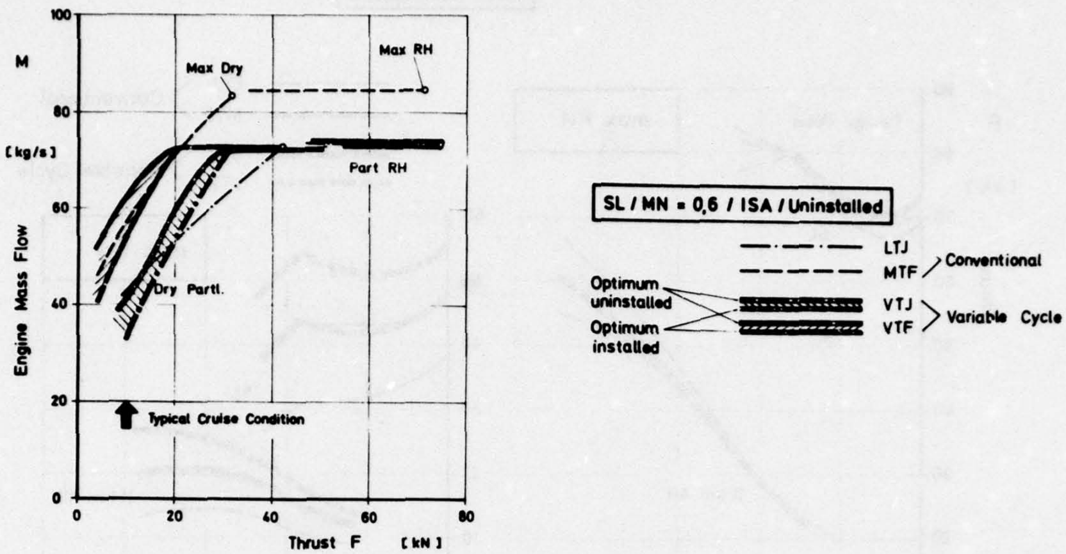

Fig. 9 Influence of Cycle Flexibility on Mass Flow Characteristics

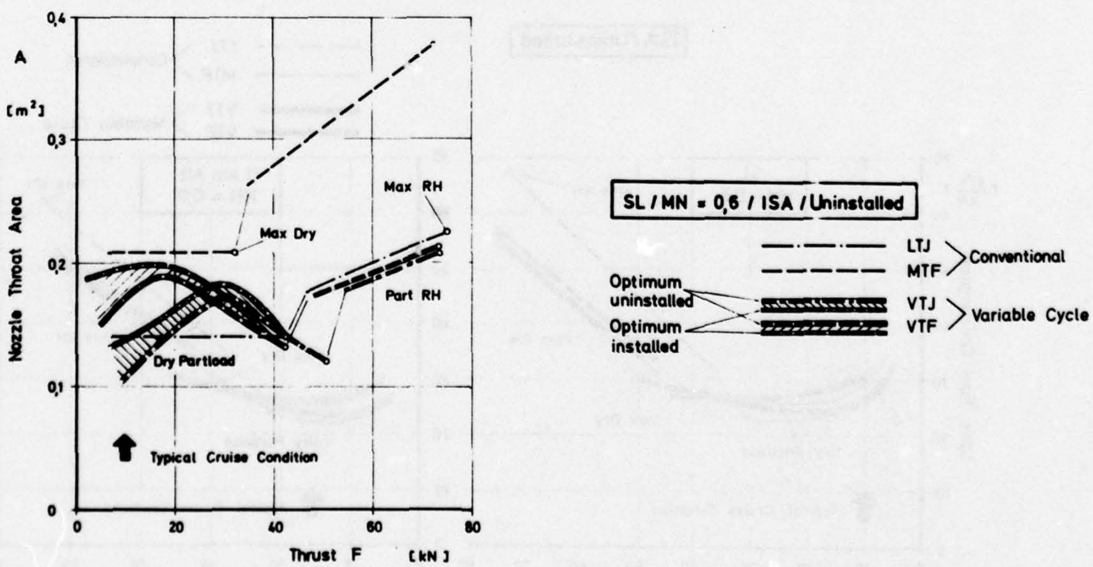

Fig.10 Influence of Cycle Flexibility on Nozzle Throat Area

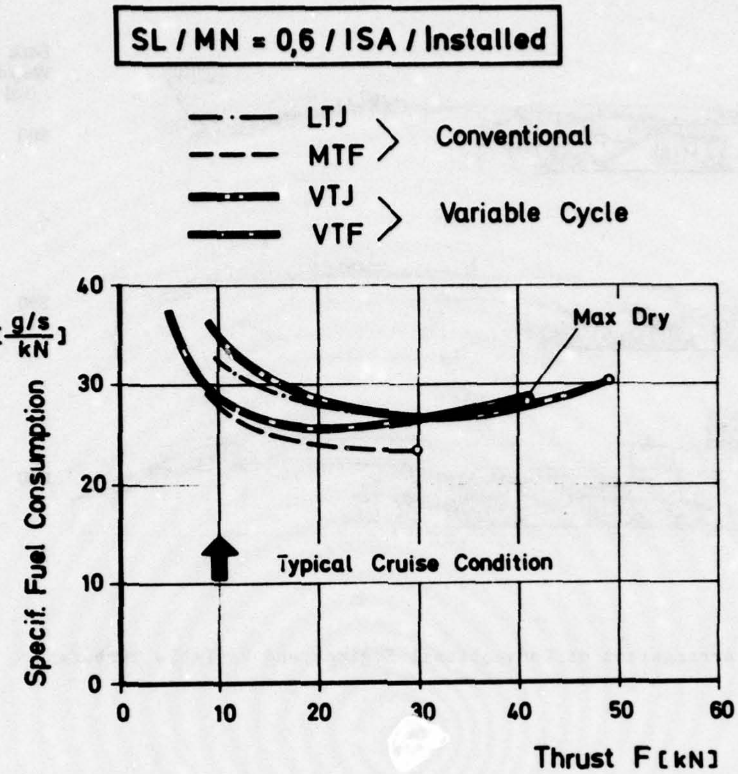

Fig.11 *Installed Dry Performance of Engines Compared*

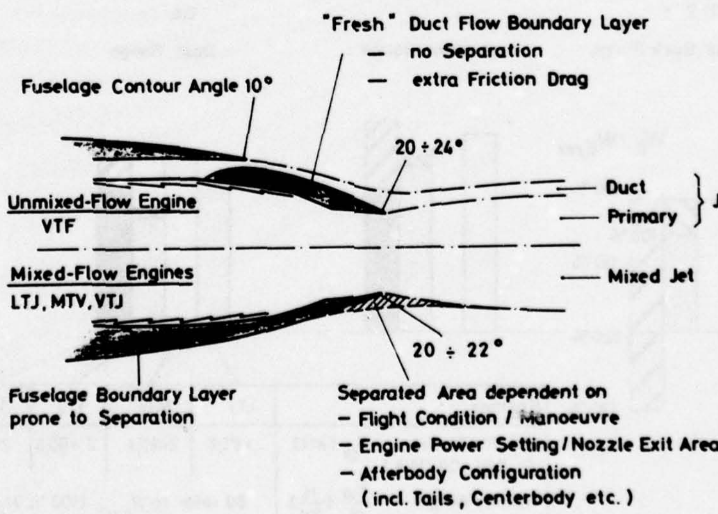

Fig.12 Subsonic Afterbody Flow Pattern (Dry Rating) for Mixed-Flow and Unmixed-Flow Engines

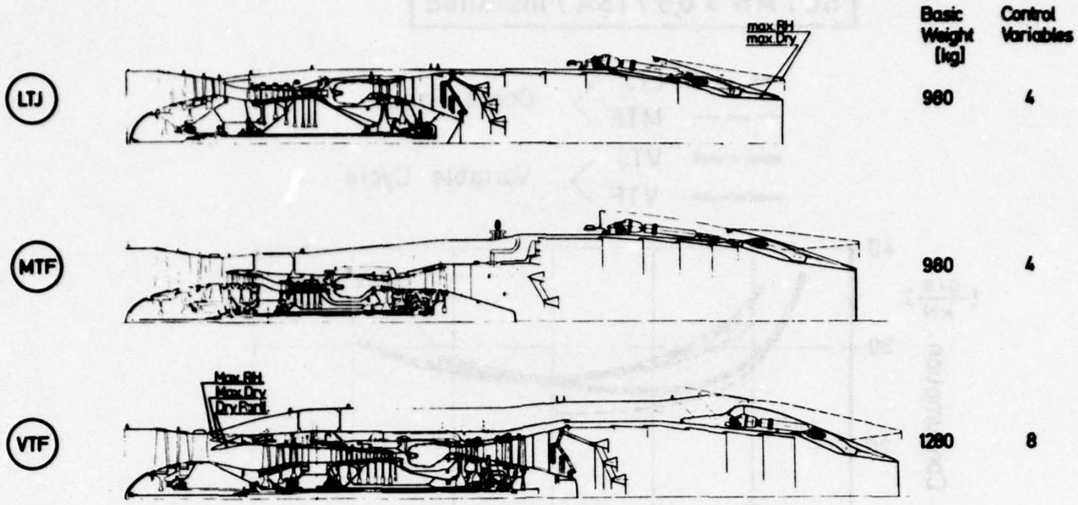

Fig.13 General Arrangement of Conventional Engines and Variable Turbofan

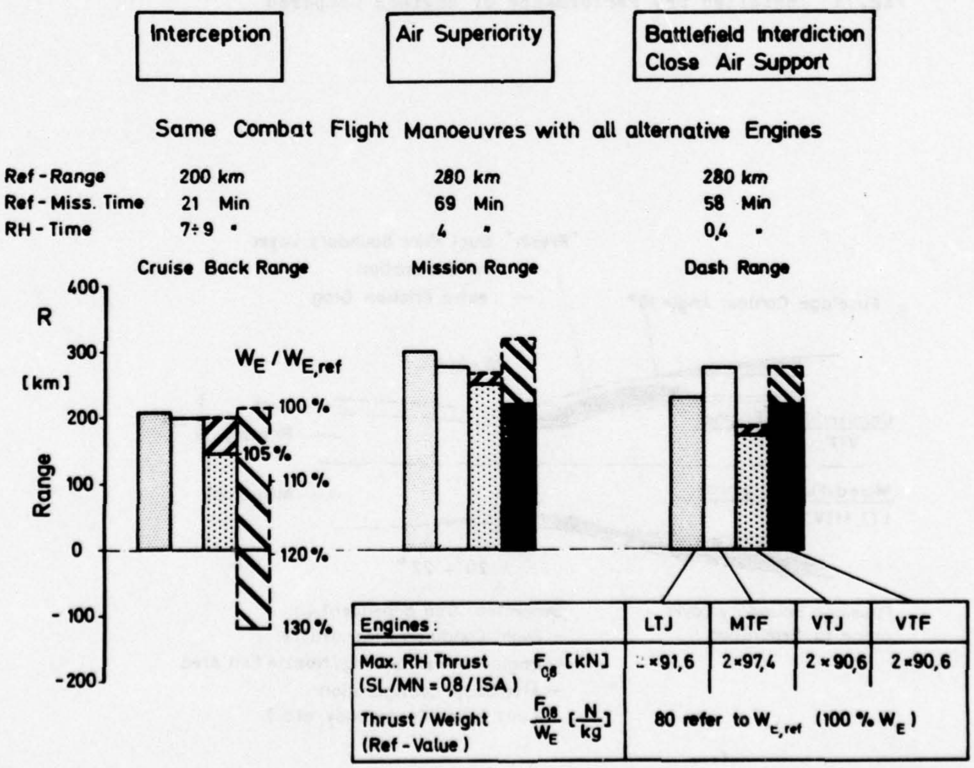

Fig.14 Influence of Alternative Engine Installation on Range of a Combat/Fighter Aircraft

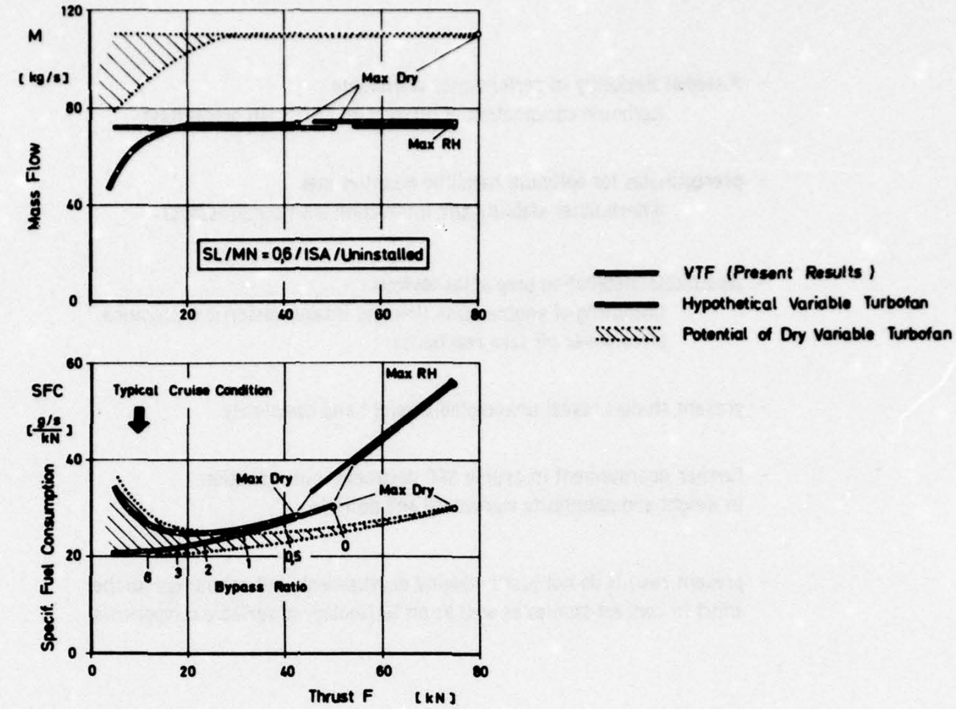

Fig.15 Performance Potential of Dry and Reheated Variable-Cycle Engines

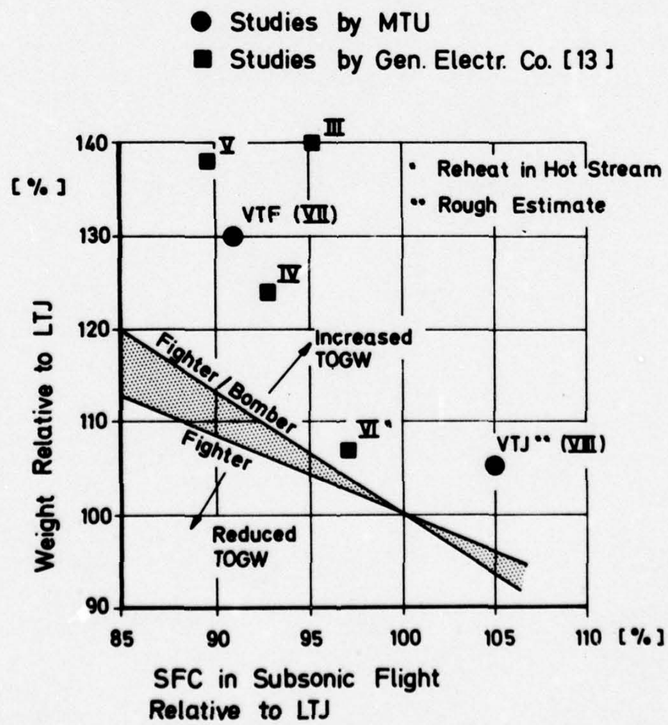

Fig.16 Break-Even Relationships for Engine Weight Increase
and Fuel Weight Reduction for Various Military Aircraft

- Aimed-at flexibility in performance achievable
  (optimum combination of turbojet and turbofan properties)

- prerequisites for optimum handling qualities met
  (afterburner stability and interaction with compressors)

- favourable installation properties obvious
  (matching of engine mass flow and intake, distortion tolerance, bleed/power off-take reactions)

- present studies reveal unacceptable weight and complexity

- further improvement in cruise SFC desirable, but reduction in weight and complexity mandatory and decisive

- present results do not justify engine development, but encourage further effort in concept studies as well as on technology of variable components

Fig.17 Conclusions

INTAKE DESIGN FOR FIGHTER AIRCRAFT

- by -

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SUMMARY

An outline is given of the requirements placed upon the intake designer. Low external drag, high intake pressure recovery, and low distortion of the engine face flow are demanded, over a wide range of aircraft manoeuvres at subsonic speeds as well as through the transonic and supersonic speed ranges. All this is to be accomplished at minimum cost, weight, and complexity.

Some of the factors involved in the design are discussed. These include intake location, blunt cowl lips and their influence on incidence range; matching of engine flow demand with intake capacity; dynamic distortion measurement; and the control system for a variable intake. An assessment of the state of the art, research requirements, and possible technological advances follows.

1.0 INTRODUCTION

This Symposium on Fighter Aircraft Design brings together specialists on many different parts of the design, so as to assemble a balanced understanding of the factors influencing the choice of the complete weapon system. One of these parts is the intake. This paper reviews the factors influencing intake design and the technology associated with intakes, not for the benefit of intake specialists but rather for the information of specialists in other fields. More detailed reviews have been presented to AGARD in the past[1,2].

This session of the Symposium is devoted to propulsion, and the question immediately arises as to whether the intake is part of the aircraft or part of the engine. While responsibility for the design and manufacture of a fighter intake lies usually within the aircraft manufacturer's contract, both the aircraft designer and the engine designer are deeply concerned to achieve a good intake. Some design compromises are necessary which trade advantages for the one against disadvantages for the other. It is therefore vital for the aircraft and engine designers to collaborate closely from the earliest stage.

2.0 *INTAKE CONFIGURATIONS*

A wide variety of configurations is to be found in current fighters, showing how the designers, weighing the advantages and disadvantages of various arrangements for their particular mission, have been driven to different conclusions.

The intake can be in the nose, as in the Lightning, or just underneath it as in the Corsair and the Fiat G91. The F16 intake is underneath but much further back. This intake is mounted slightly away from the fuselage so that the fuselage boundary layer does not enter the intake but is diverted back into the free stream. Diverters are commonly found also on side intakes such as the Jaguar and YF17, each adopting a different shape of intake. The Harrier on the other hand has no diverter but has a boundary-layer-bleed. Auxiliary intake doors (which are sucked open at low speed) can also be seen, for example in Harrier and Jaguar. Fighters designed for higher supersonic speeds have more elaborate intakes with diverters and with variable ramps or translating spikes, as for example the Starfighter, Phantom, Mirage, F14, F15, and Tornado. The F15 has the additional flexibility of rotating the whole front section. The MIG 25 variable ramp intake has a hinged lower cowl lip (as had the B-1 variable geometry intake).

Why are there all these variations?

3.0 SUMMARY OF REQUIREMENTS

In this Section, the requirements placed upon the intake designer are summarised. In subsequent Sections, each requirement and the technology available to meet it is examined in more detail.

The fundamental performance requirements are to supply the airflow demand of the engines at the highest possible pressure recovery and with minimum drag on the airframe. These (together with equivalent exhaust system requirements) ensure maximum available thrust and propulsive efficiency, providing the highest possible speed and endurance.

The third aerodynamic requirement is to provide the engines with as steady and as uniform an airflow as possible. Although distorted airflows can lose some performance, the main reasons for minimising distortion are to avoid surging the engines and to avoid exciting severe blade vibration. This is relatively easy in straight-and-level flight, but of course a fighter must be able to manoeuvre violently over a wide speed range. Figure 1 shows typical incidence and yaw envelopes[3,4]. A number of factors can limit the manoeuvring envelope - structural, physiological, maximum lift coefficient, thrust over drag margin - but engine surge initiated by intake airflow distortion is another potentially significant factor. Engine surge provoked by airflow disturbances following the firing of guns, rockets, or missiles must also be avoided.

Minimum weight and cost of development, production, and maintenance are obvious requirements, militating against the more complex variable geometry designs.

Flight safety is another important factor influencing intake design. In the case of a variable intake, a safe mechanism and control system must be chosen, but even with a fixed intake two factors must be considered. One is foreign object damage. Fighter aircraft are very prone to foreign object damage, and if the intake is too near the ground or just behind the nose wheels, it will suck in anything lying on the runway. The other factor is that of radar reflection; some designs of intake reflect radar beams efficiently and make detection much easier.

It will already be evident that the final design of intake must be a compromise. Better performance and lower distortion must be balanced against the higher weight and cost of variable geometry; higher drag may be balanced against higher pressure recovery. That is why the airframe and engine designers must collaborate from the start, and why intake configurations vary so much.

4.0 DETAILED REQUIREMENTS

4.1 AIRFLOW MATCHING

The intake must supply the engine with the flow it needs. This presents no problem for a subsonic intake at subsonic speed. A supersonic intake, on the other hand, has an optimum flow, departures from which can lose efficiency, increase drag, or lead to instability; consequently the matching needs careful consideration.

Figure 2 shows diagrammatically the variation in mass flow✓ required by the engine, and that supplied by the intake. This shows that a sharp-edged supersonic intake (the form most efficient for sustained cruise at Mach 2) must have variable geometry for low speeds, such as auxiliary doors, whereas a blunt lip intake matches better. It also shows that at high Mach numbers at altitude (and not such high Mach numbers at sea level) a variable geometry engine would potentially enable better matching to be achieved and hence improve high speed performance.

4.2 INTAKE DRAG

It is convenient to define the standard net engine thrust as the engine stream thrust leaving the nozzle less the engine stream momentum far upstream of the aircraft (Figure 3):-

$$F_{NET} = F_2 - F_0$$

where the stream force F = momentum + excess pressure

= flow $\times$ velocity + area $\times$ gauge pressure

This net thrust can be written

$$F_{NET} = (F_2 - F_1) - (F_0 - F_1)$$

= thrust provided by engine

- momentum lost approaching intake

This second term would be fully offset in an isentropic flow by the "cowl suction" around the intake surfaces, but in a real flow it is not completely offset and the difference is called "spillage drag". The value of spillage drag naturally depends on the particular intake and is measured in a wind tunnel. Then, knowing the aircraft drag, the spillage drag can be expressed as a fraction of the drag of the complete aircraft. Figure 4 shows typical figures, illustrating how significant drag can be caused by spilling overboard flow which the intake could swallow but the engine does not want.

In order to achieve high pressure recovery and low distortion and to delay the onset of the instability known as "buzz", it is found necessary to avoid ingesting the fuselage boundary layer. This is done by mounting the intake a finite distance from the fuselage and diverting the boundary layer flowing into the gap between them. This naturally incurs a drag penalty due to friction and (at supersonic speeds) due to wave drag. This drag depends on the shape of the diverter and is of the order of 2.5 per cent of total aircraft zero-lift drag.

4.3 PRESSURE RECOVERY

The intake pressure recovery is defined as

$$\frac{\text{engine face mean total pressure}}{\text{total pressure that an adiabatic isentropic compression would achieve}}$$

This is less than unity because of friction, shock waves (in supersonic flight) and in some circumstances flow separations.

✓ "Corrected mass flow" denotes $\dfrac{\text{mass flow}}{\text{pressure (atm)}}\sqrt{\dfrac{\text{temperature }(^{\circ}\text{K})}{288}}$

A simple fixed-geometry pitot intake can be designed for low pressure loss (friction only) in subsonic level flight, but in supersonic flight a normal shock wave forms ahead of it. Figure 5 shows the pressure recovery obtainable in principle from such an intake. At higher speeds, a wedge or spike is desirable to generate one or more oblique shocks before the final normal shock, to improve the pressure recovery as shown. An increase in top speed from Mach 1.8 to Mach 2.2 makes such an arrangement essential.

Figure 6 shows diagrammatically how the recovery of a ramp intake varies as the intake flow is changed. There is an optimum flow; just above it the intake chokes; somewhat below it the flow becomes unstable and a fairly violent oscillation known as intake buzz occurs. The total range is quite narrow, less than 10 per cent at Mach 2. Therefore, to accommodate desired flow changes through the engine as flight speed, altitude, and climatic conditions change, this form of intake is usually provided with throat area and shock pattern variation either by moving the ramps (in the two-dimensional or wedge intake) or by translating the central plug axially (in the axisymmetric or spike intake). Figure 7 shows that the F111E intake achieves high recovery over a wide range by an expanding cone and translating spike[5].

In assessing the relative importance of intake drag and pressure recovery, it is useful to remember that (since F_2 is two or three times F_{NET}) 1 per cent of pressure recovery is worth 2 or 3 per cent of drag.

If the intake is optimised for supersonic cruise, the leading edges of all sides of the intake are sharp (as in Concorde, for example). If, on the other hand, a high pressure recovery and high sustained manoeuvrability are needed in the transonic range, the cowl lip needs to be blunt. Figure 8 illustrates this for the F15 (Reference 6).

After entering the intake, and passing through the shock pattern, further diffusion is needed down to the Mach number of about 0.5 required by the engine. This is accomplished in the duct joining the intake to the engine, which in many aircraft has a complicated (and sometimes fairly sharp) double bend. To achieve high pressure recovery and uniform engine face flow it is necessary in most cases to employ boundary layer control, in the form of bleed, blowing, or vortex generators, within the supersonic portion of the intake if not the subsonic portion.

The design of the duct is constrained by the aircraft structure and undercarriage, and in general it is so closely integrated with the airframe that once chosen it cannot be enlarged. Now it is well-known that an aircraft tends to increase in weight during its development history and demands arise for increased engine thrust to maintain or enhance performance. Increased thrust can only be achieved by higher engine temperatures or by increased airflow. So the prudent designer allows for possible future increases in airflow in the initial intake and diffuser design.

A particularly difficult intake problem is posed by a VTOL fighter with vectored thrust of the Harrier configuration. The engine has to be near the centre of gravity of the aircraft, so is found much further forward than in conventional fighters. This allows only a short length intake, which if it is a side intake implies a very sharp double bend ahead of the compressor face.

4.4 DISTORTION

If the airflow reaching the engine is unsteady or non-uniform, its surge margin is reduced, and in extreme cases the engine will surge and flame-out. Surge problems caused by this intake "distortion" have been encountered in several aircraft, and experience with an early version of the F111 led to the imposition of stringent specification requirements on subsequent military aircraft to avoid trouble in future.

The whole subject is reviewed in Reference 7. Flow distortion and unsteadiness can arise from three sources:-

1. Non-uniform total pressures caused by shock wave intersections within the capture flow, which can occur at off-design conditions.

2. Flow separations from the intake lips or within the diffuser, which can occur at high incidence or sideslip or flight Mach number, or running with cross-wind on the ground.

3. External disturbances caused by firing guns, missiles or rockets near the intake, or reingesting the hot gases from the downward jets of a VTOL fighter near the ground.

In general, high distortion levels are associated with low pressure recovery conditions, since both arise for similar reasons. Naturally, there is little distortion from a properly designed intake, even at supersonic speeds, in level flight; the high distortion arises during extreme manoeuvres (or weapon firing). The main danger is that operational limitations imposed by surging might only be revealed during service trials when it is too late to make major changes.

The first difficulty lies in defining some quantitative parameter (known as a "distortion index") which is uniquely related to loss of surge margin. Prolonged effort by the engine companies has evolved a range of indices ($DC\theta$, KD, K_θ, KA_2 etc). Each of these indices has been related empirically to loss of surge margin on a particular engine with reasonable success.

Methods have been evolved, and are now applied in routine fashion, for measuring distortion indices, for testing engine sensitivity to them, and hence for estimating at an early stage of development the likelihood of distortion-induced engine instability.

It is necessary to measure "instantaneous" or dynamic distortion, because experience has shown that distortion varies in a random fashion, with peaks typically double the time-average value, and that such peaks can surge the engine even if they last only for one engine revolution (of the order of 5 ms). Therefore these methods involve mounting forty or fifty high-frequency-response pressure transducers within

the intake, recording their output, and processing them by computer to generate a millisecond-by-millisecond history of the distortion level. Figure 9 shows an example. The lower trace shows the highly erratic distortion index and the upper trace shows the average inlet pressure. About 25 ms after a peak (labelled A) in the distortion, the engine surges (B) as revealed by the upper trace.

Figure 10 shows the variation of distortion index for the F111E in level flight[5]. Comparison between the peak dynamic value and the time-average value illustrates the need for the peaks to be measured. In general, by comparison with Figure 7, it will be seen that high distortion and low recovery usually coincide. The high distortion level when standing on the runway is noticeable. (Bleed valve operation on the TF30 engine can more than double the engine tolerance, if necessary.)

These methods do not cover the case of missile and rocket firing, however. If the aircraft and missile trajectories are such that the intake ingests the rocket exhaust, a burst of very hot gas enters the engine and can cause surge. Figure 11 from Reference 8 shows an example. It is possible, as shown in that paper, to simulate this in an engine test facility. Disturbances from guns can sometimes be eliminated by shielding, but temperature peaks resulting from missiles and rockets cannot be eliminated by the intake, and the engine control system has to use a signal from the firing button to provide a momentary fuel flow reduction to avert surge.

4.5 DESIGN FOR HIGH INCIDENCE OPERATION

As has been pointed out, manoeuvrability may be limited by engine surge caused by excessive incidence or sideslip. The key factor is the local angle-of-attack of the airflow around the front edges of the intake. If that angle is too great for the lip to turn the flow into the intake without local separation, a distorted and unsteady flow will go into the engine. To avoid this as far as possible, especially in the transonic speed range, the intake lip is blunted or rounded as shown in Figure 8 for the F15. This lip, and the Tornado cowl lip, are quite blunt compared with that of Concorde optimised for similar flight speeds without the manoeuvre requirement.

A complementary design approach is to locate the intake in such a position on the aircraft that either the fuselage, a strake, or the wings shield the intake so that it experiences a local angle of attack much less than aircraft overall angle of attack. A major research programme known as "Tailormate" was mounted in the USA to find, by wind tunnel model tests, where best to locate the intake on a fighter to shield it from incidence and yaw changes as far as possible. It was found that the shape of the fuselage is also a significant factor. As an example, Figure 12 (Reference 9) shows the effect of fuselage shielding on side intakes and Figure 13 (Reference 4) on an F-16 underbelly intake, which can be seen to be particularly good at incidence.

Fixed geometries can only achieve a limited incidence range. The solution adopted in the F-15 (Reference 6) was to arrange for the whole front portion to rotate into the flow direction; the effect of this is shown in Figure 14.

There is much evidence that thrust vectoring in flight, or manoeuvres like the F-14's violent jink[3], can enhance combat effectiveness. It is evident that such manoeuvres impose higher incidences on the intake than normal, and they require more radical action to avoid lip separations. Cawthon[10] explored a range of possibilities and showed their effectiveness. His tests extended to 56° incidence. Figure 15 summarises his results; the improved pressure recoveries shown are accompanied by reduced distortion levels.

4.6 FLIGHT SAFETY

The ingestion of foreign objects into the engine is a hazard to the aircraft. The Royal Air Force alone estimates that the costs incurred as a result of foreign object damage amount to around £12M every year.

Birds represent one hazard. From time to time intake guards have been proposed but they have always in the past been judged too heavy, or operationally unsuccessful in that when retracted they neatly deposit the bird into the engine!

Stones, rivets, and even spanners left in the intake or sucked off the ground are another hazard. Vigorous steps are taken by all Air Forces to avoid carelessness in leaving these objects in the intake or on the ground, but there has been some controversy over how a stone can get from the ground into an intake. Tests undertaken by NGTE at model scale[11] and by the Royal Air Force have shown that although vortices can form between the ground and the intake which attempt (like a meteorological tornado) to suck objects off the ground, they only succeed when the intake is very near the ground, of the order of 1.3 intake diameters or less clear of the ground. In designing an aircraft with an underbelly intake (like the F-16) it is necessary to maintain adequate ground clearance in this respect. The investigation also demonstrated that a jet sprays any stones lying on the runway up into the air behind the aircraft, and this appears to provide the main hazard to following aircraft or to nearby parked aircraft, and one the fighter intake designer can do nothing about.

Of course, the moving parts of a variable intake also have been known to come loose, and it is important to check the structural integrity under the worst conditions. In the case of Tornado, for example, the shock loading stresses were checked by instrumenting an intake mounted in the altitude test facility at NGTE, and deliberately surging the engine.

4.7 COST AND WEIGHT

The major factor controlling both cost and weight is the degree of variability. Obviously a fixed geometry intake is lighter, and cheaper in development, production, and maintenance. It must not be forgotten, either, that most forms of variability require a control system which brings further expense. As a rough guide, the adoption of a controlled variable intake can add around 3 per cent to the cost of the aircraft.

5.0 INTAKE CONTROL SYSTEMS

A number of intakes have auxiliary doors (eg Harrier and Jaguar) which open at low speed to improve the intake flow and matching. Some are spring loaded to open when the pressure difference across them reverses; they are thus self-positioning and require no actuation by a control system. On the other hand, a variable ramp intake or translating centrebody intake requires actuation.

The Tornado intake control[12] consists of an actuator driving the ramps to a position such that the bleed pressure behind the ramps is a scheduled function of intake total pressure, flight Mach number, and aircraft incidence. The operation is controlled by a digital electronic computer with appropriate fault detection devices and manual override. The F-14A intake control[13] is similar but controls as a function of engine fan speed. Experience of supersonic intake control systems has of course also been accumulated on Concorde.

A noticeable feature of all these systems except Concorde's is that the intake control is independent of the engine control; the concept of a "powerplant control" has not reached production. An NGTE study has suggested that so little dynamic interaction exists between engine and intake that little performance or handling advantage can be gained by such a unified control. Flight tests on an F111E (References 5, 14), on the other hand have demonstrated the value of feeding intake distortion information to the engine control. If a digital computer is used for engine control, as appears very likely in any future fighter aircraft, it could also handle any intake variable and hence reduce the cost penalty incurred in providing a separate variable intake control; this cost benefit has been estimated[15] as 40 per cent on purchase cost and 25 per cent on maintenance cost of the control system.

6.0 AN ASSESSMENT OF THE STATE OF THE ART

6.1 MACH 2 TO 3

Extensive experience has been accumulated in the United States and also in Europe of the design of the variable intakes required in this flight speed range. While the amount of wind tunnel testing required to develop a good variable intake could certainly be reduced by continuing improvements to two-dimensional and three-dimensional transonic and supersonic flow field calculation methods, and also by a deeper understanding of optimum boundary layer control techniques, there is little doubt that the design of a new fighter intake for this speed range could be undertaken with great confidence.

6.2 MACH 1 TO 2

This is generally regarded as a more likely requirement, and after due consideration of variable intakes, the choice of a fixed geometry intake for F16, 17, 18 (and B1) with a slight loss of top speed and of performance above around Mach 1.8, is very significant. As at higher speeds, the evolution of an optimum design involves extensive wind tunnel tests and could be improved by advanced three-dimensional flow field computations. These could also help to interpret the immense quantity of data accumulated in the United States "Tailormate" programme investigating the optimum aircraft configuration. The detailed shape of the cowls and of the internal diffuser needs attention. The main problem is selecting the right compromise between high-speed and low-speed requirements, and in this respect the designers have not perhaps reached the top of the learning curve.

6.3 MACH 0 TO 1

While future fighters will normally have supersonic top speed, the air-to-air-combat region is subsonic and here manoeuvrability is the prime consideration. This is likely, in the author's view, to lead to thrust vectoring or thrust modulation in some form, which will impose wide incidence limit requirements on the intake. To meet them, more experimental research will be needed on high-incidence cowl lips (even if the best-shielded location of the intake is selected). Either variable-geometry (drooping lips or leading-edge slats) or boundary-layer-control by local suction or blowing may well be needed, and research to provide them should be undertaken now.

7.0 TECHNOLOGICAL ADVANCES

What improvements in fighter aircraft intakes can be envisaged other than by steady attention to detail and rigorous wind tunnel testing of current concepts?

One evident need is for some device which will enable the intake to be optimised for subsonic and transonic flight, as a pitot intake, and yet incorporate some relatively simple way of generating an oblique shock at supersonic speeds to avoid the high normal shock loss (Figure 5). Goldsmith at the Royal Aircraft Establishment has suggested a possible way of doing so, in the form of a retractable step with appropriate bleed slot behind it. Reports of his initial investigations are yet to be published, but some device of this kind may prove useful in a new fighter.

Another new concept, referred to in Section 5.0, is the exploitation of the immense flexibility and computing power and relatively low cost of emerging digital microprocessors to control intake variables in concert with engine variables. In particular, the in-flight detection of sudden intake flow disturbances (due to weapon firing or extreme aircraft manoeuvres) at a point far enough upstream of the engine to allow control action on engine fuel flow, variable stators, or bleeds, to avert threatened surge offers a real prospect of relieving the pilot of one of his handling worries.

A third technological advance can be expected, and is well worth striving for, in dynamic distortion measurement and analysis. Following the F111 problems referred to in Section 4.4, immense efforts in manpower and money were devoted to evolving expensive equipment in the form of high frequency response rakes, recorders, and computers, all now insisted upon by procurement authorities. Already, the early analysis methods have been speeded up to nearly real-time and no longer need be expensive, but if the research efforts now in progress[16,17] on the statistical properties of these unsteady flows can be

successfully completed, the need for such elaborate tests can be entirely swept away within ten years.

8.0 CONCLUDING REMARKS

The correct choice of engine intake can greatly enhance the performance and manoeuvrability of a fighter aircraft. Most of the technology required is already available, but current research will evolve not only some improvements but also better analytical techniques which can shorten development timescales and reduce their cost. The main feature of the design is however the need to compromise between the sometimes conflicting requirements of different parts of the mission, namely getting to the combat zone and the combat itself. Therefore, please will the procurement authorities specify the mission rules as clearly as possible and allow the intake designers time to evolve the best compromise in their wind tunnels before freezing the aircraft design.

It is also essential to test the intake-engine combination in an altitude test cell such as NGTE Cell 4 before flight, both to check the intake control system and to assess intake-engine compatibility (that is, the surge margin of the engine behind the intake).

Secondly, to provide development potential in the powerplant (higher airflows), will the aircraft designers please not build the intake front section into the main aircraft structure. Modifications must be possible. The aircraft and engine companies have historically had quite different approaches to modifications. During engine development, continuous modifications provide the normal way of reaching specification targets, whereas aircraft modifications are viewed with reluctance. It is the author's view that the intake should be treated in an intermediate fashion between these extremes.

The final remark follows from this and may well be controversial. This paper and the preceding paper have discussed the need for and the practicability of variable geometry features in the intake and the engine to enable the wide mission requirements of a fighter to be met. It is the author's opinion that, from the point of view of engineering sense and cost-effectiveness, it is much better to install the variability in a relatively cold region like the intake than in the alternative location, the very hot region of the turbine nozzle guide vanes.

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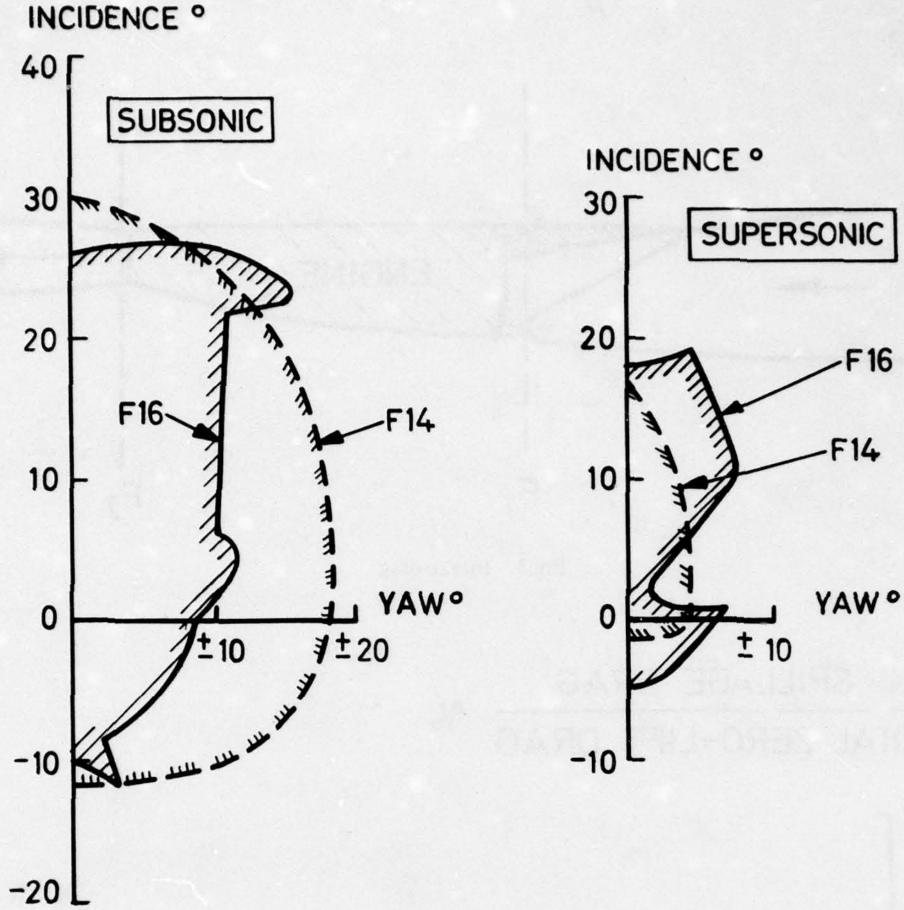

Fig.1 Manoeuvre envelope

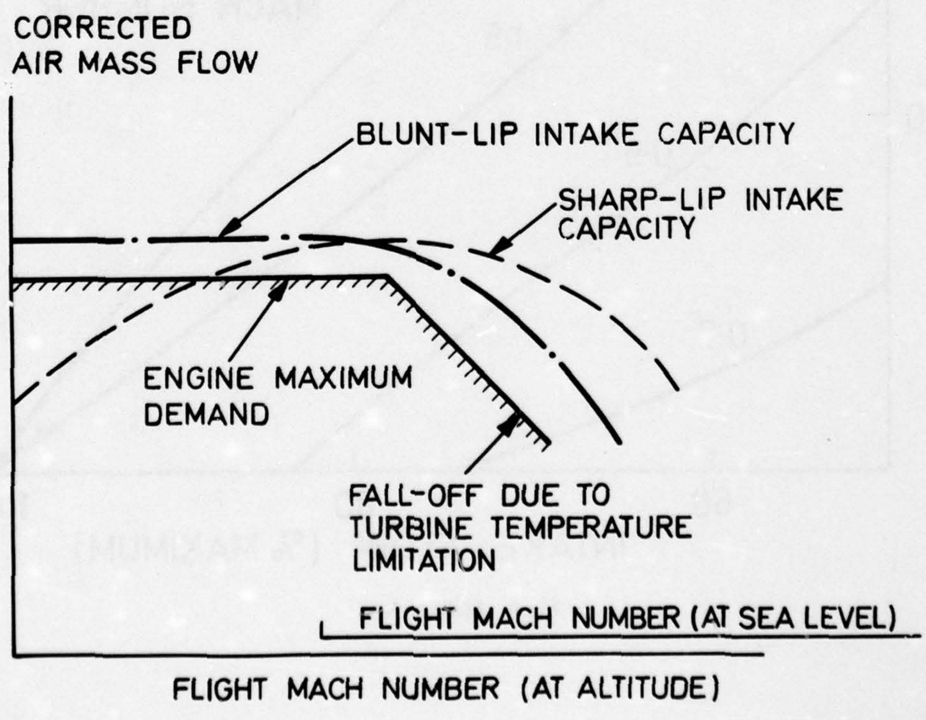

Fig.2 Airflow matching diagram

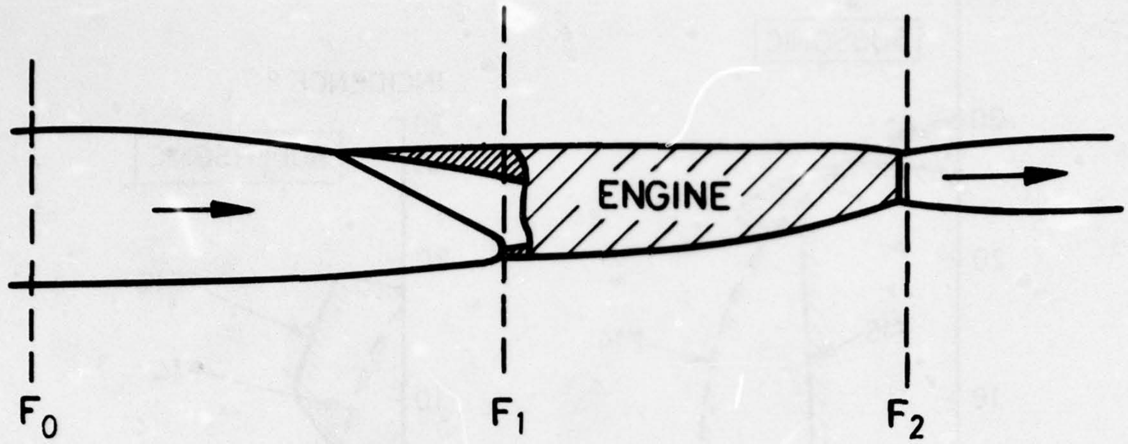

Fig.3 Intake drag

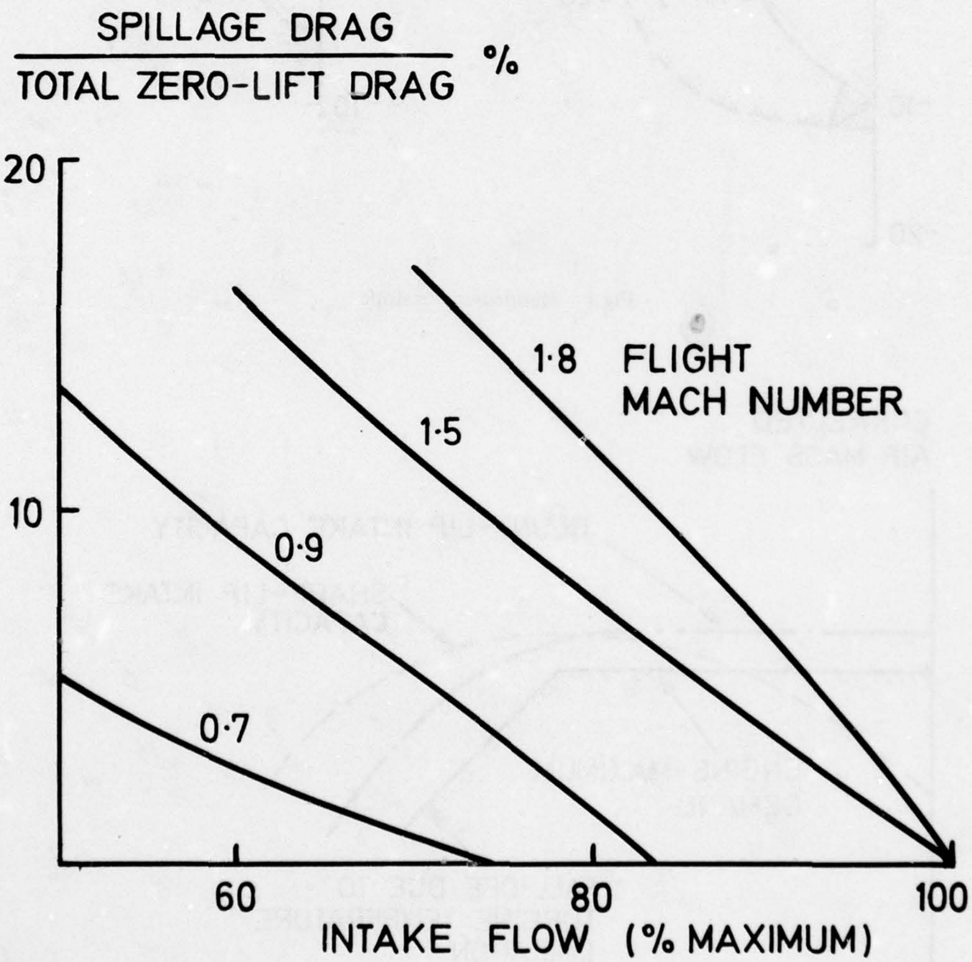

Fig.4 Spillage drag

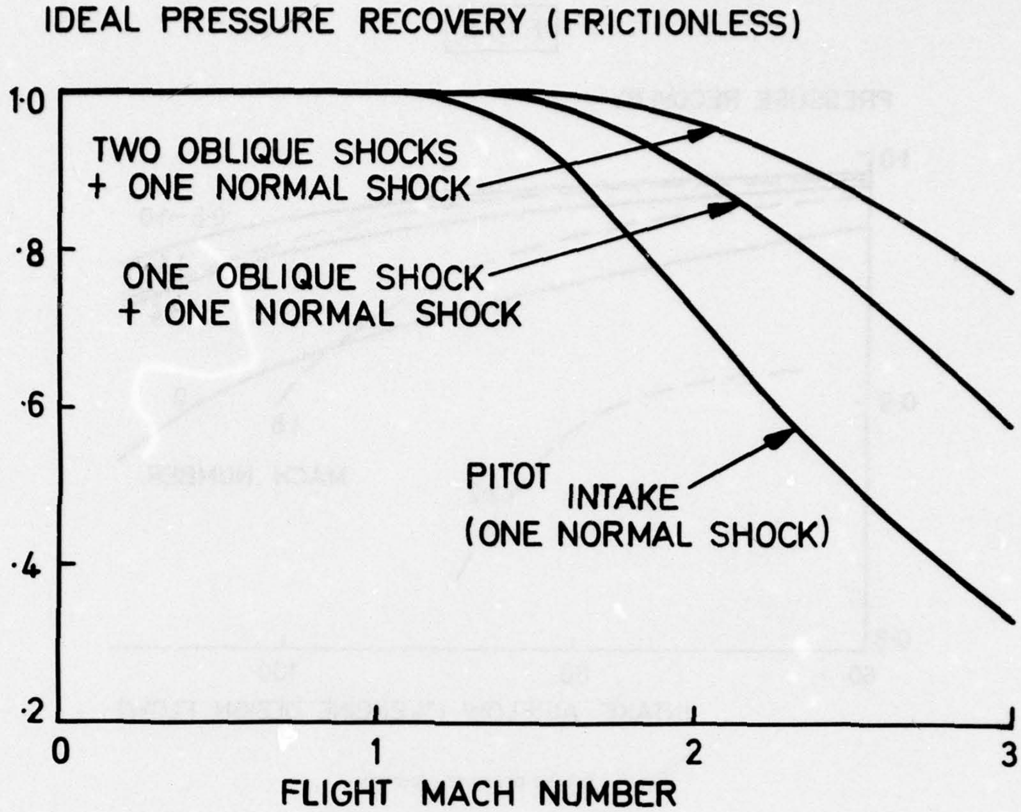

Fig.5 Ideal pressure recovery

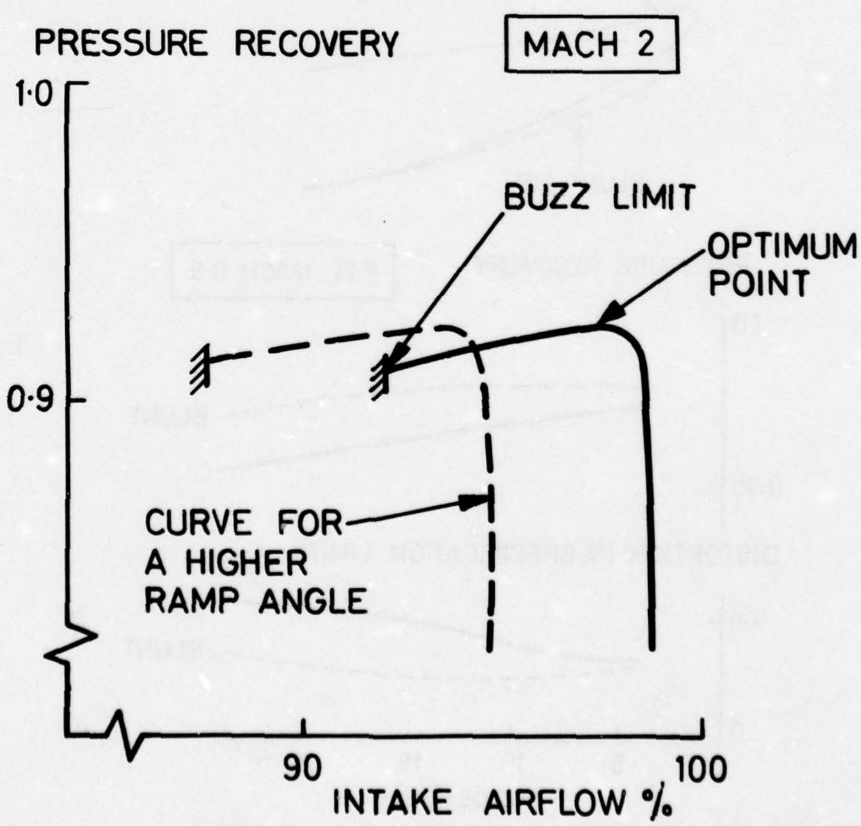

Fig.6 Performance of a ramp-type supersonic intake

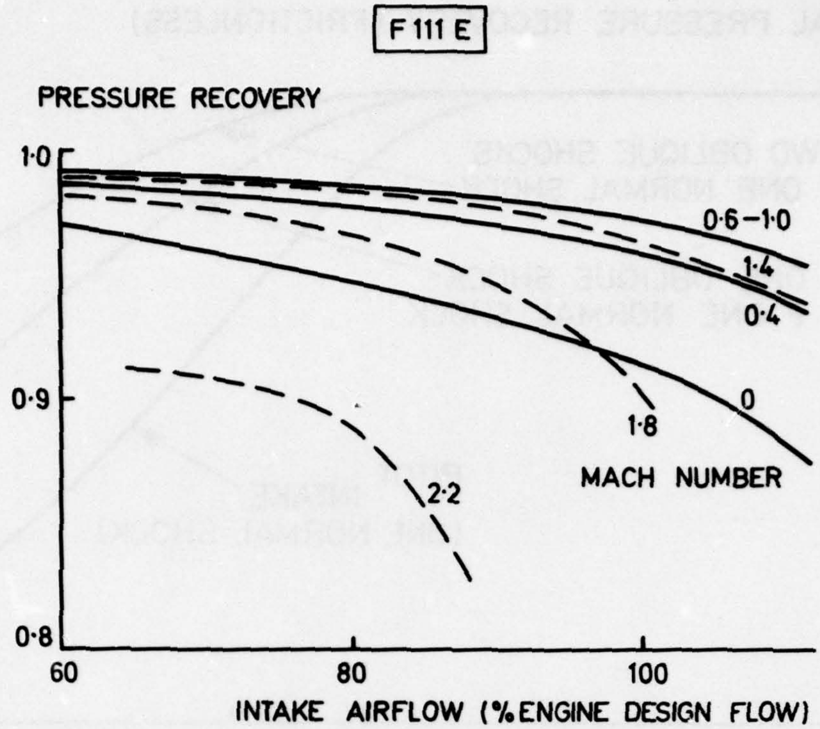

Fig.7 Intake pressure recovery

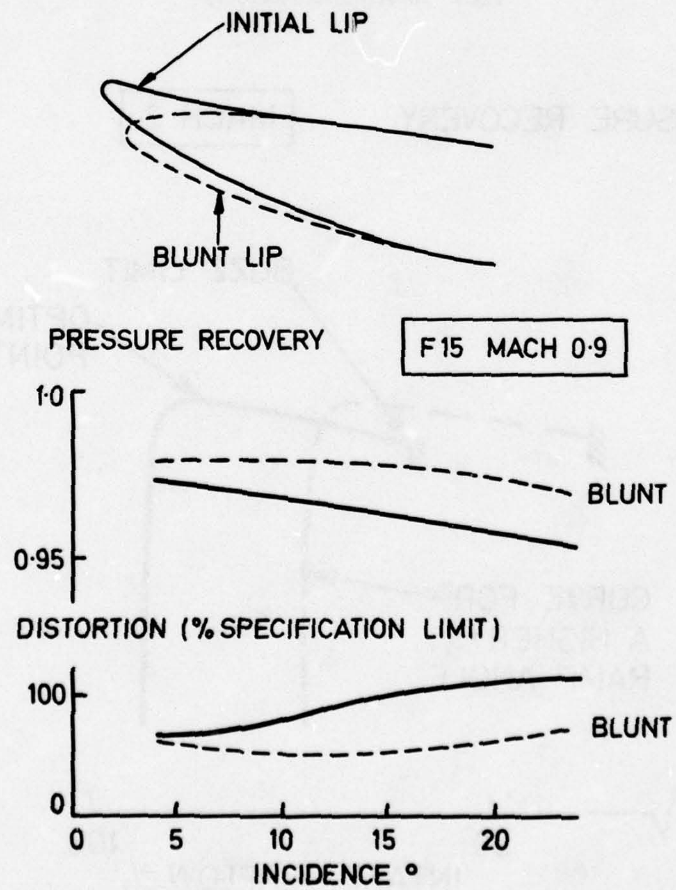

Fig.8 Effect of blunt lip

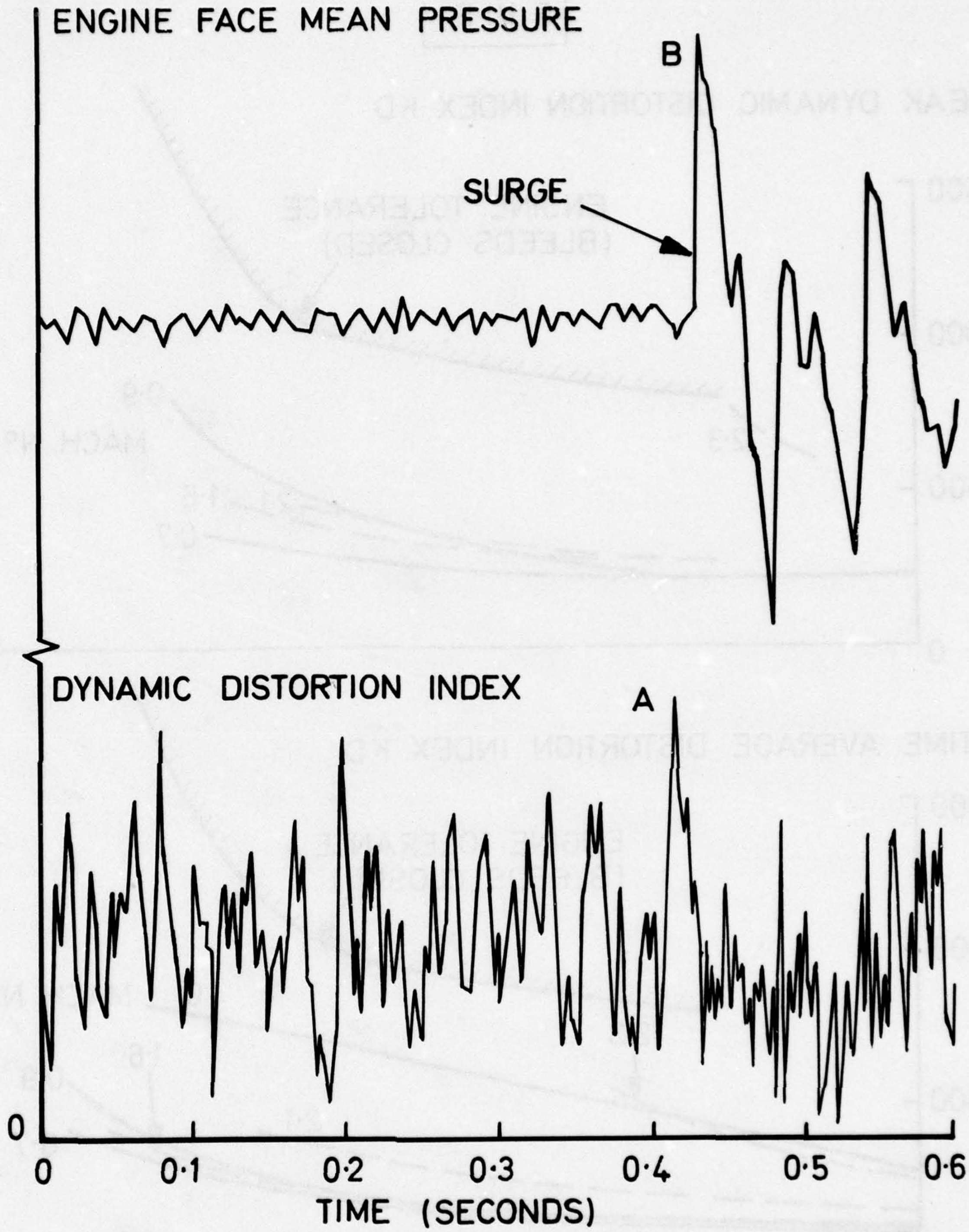

Fig.9 Dynamic distortion

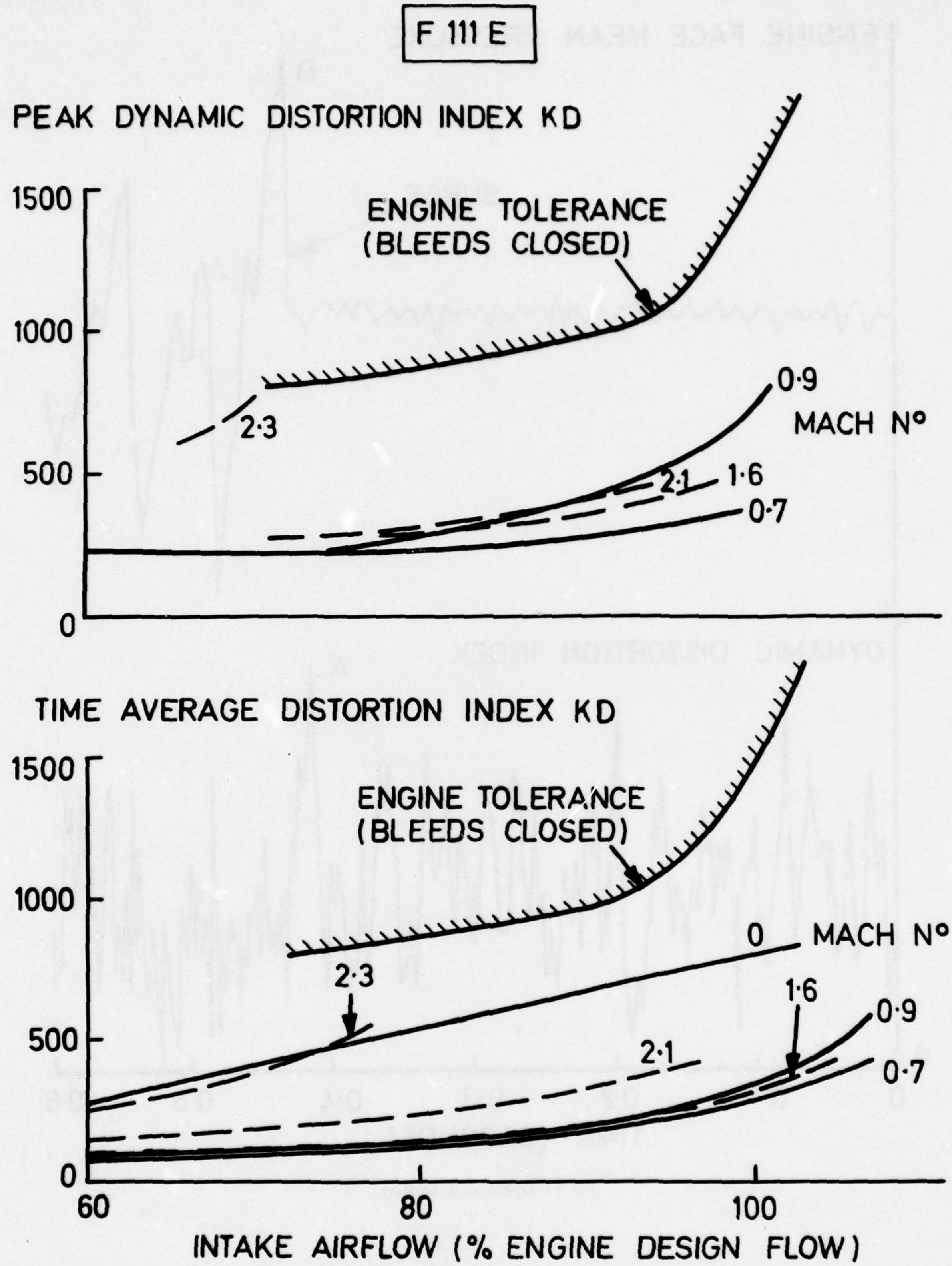

Fig.10 Intake distortion

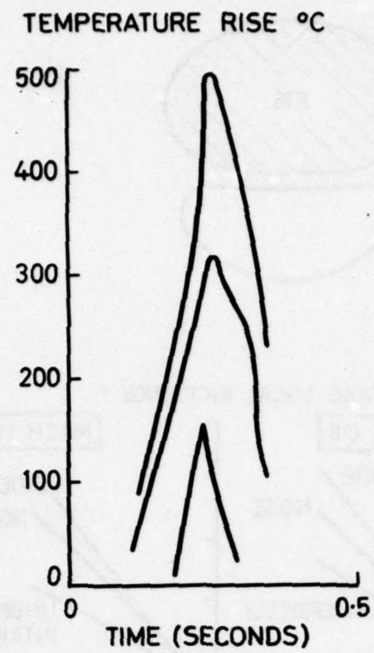

Fig.11 Temperature rise due to rocket firing

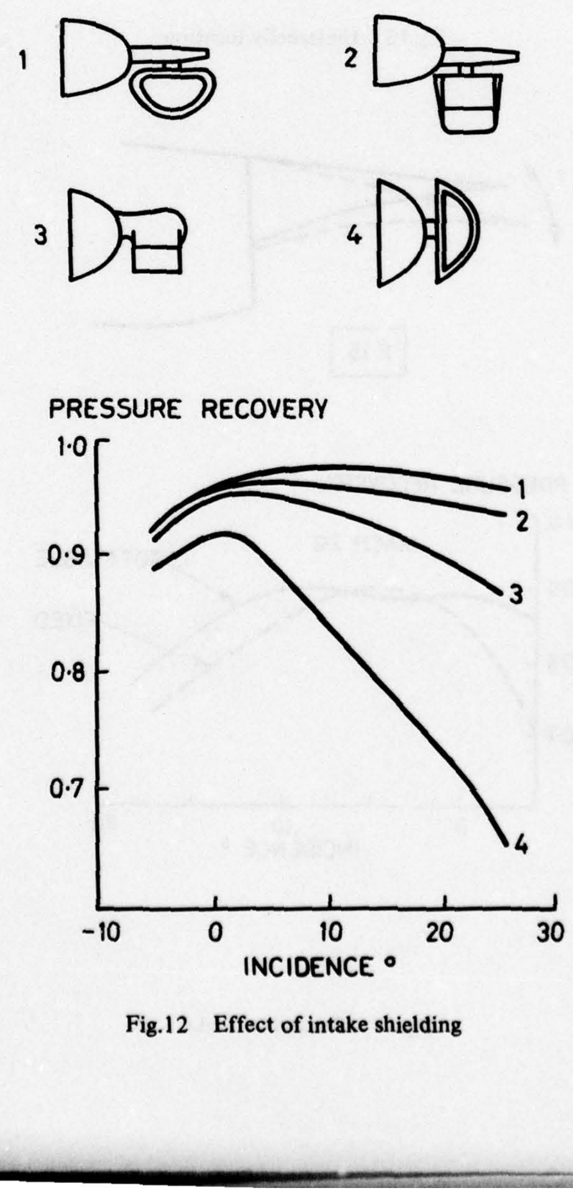

Fig.12 Effect of intake shielding

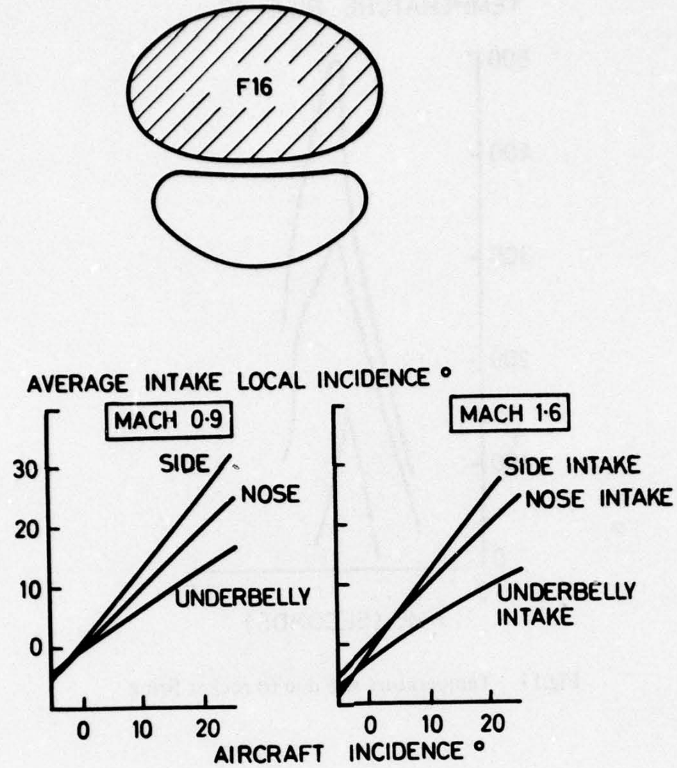

Fig.13 Underbelly location

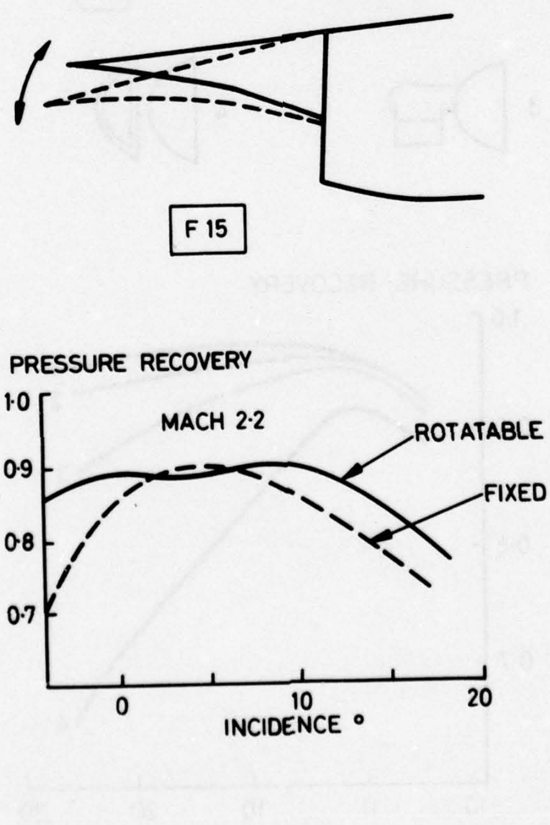

Fig.14 Rotatable intake

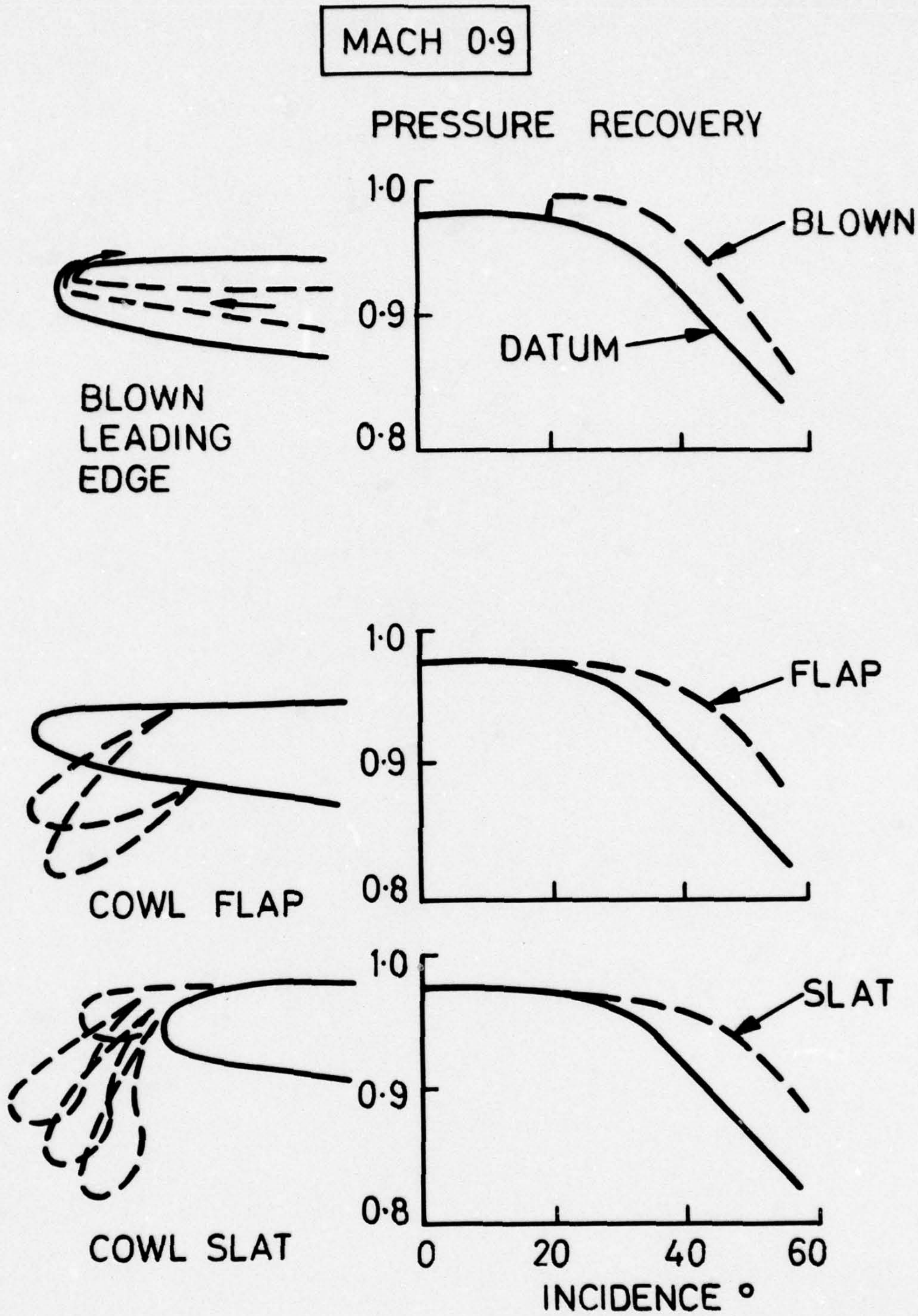

Fig.15 High incidence cowl lips

ADVANCED NOZZLE TECHNOLOGY

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SUMMARY

This paper contains a report on some of the exhaust concepts being developed as design options for use on the turbine engines which will power advanced fighter aircraft. Nozzle configurations which are not axisymmetric receive the major portion of the attention. The benefits which might be obtained through the use of these types of nozzles are discussed and three general classes of nonaxisymmetric nozzles are described. If these types of nozzles are to provide competitive design options, it is important to attain specified levels of internal nozzle performance, cooling effectiveness, and structural integrity at minimum weight. The results of recent work in these areas form the basis for establishing the level of installation benefits which must be obtained to make these nozzles competitive with the more conventional axisymmetric configurations. The use of thrust vectoring to obtain improved normal load factor capability in an air superiority fighter was examined. Variations in thrust vector schedule, aircraft wing loading and thrust loading were examined to determine the sensitivity of the potential gain to the values of these parameters. An example is used to illustrate the point that certain non-axisymmetric nozzle types and thrust vectoring schemes introduce an additional degree of complexity into the procedures necessary to properly account for the applied forces. The paper closes with a short discussion of the further development required before these concepts can be considered as design options of acceptable risk.

LIST OF SYMBOLS

AR	Nozzle Duct Aspect Ratio
A_9	Nozzle Exit Area
C_v	Gross Thrust Coefficient
$D_{a/c}$	Drag-Aircraft Wing, Fuselage, and Tail Surfaces
D_{can}	Drag-Canard
D_{ram}	Drag-Ram
F_g	Gross Thrust
Fn	Net Thrust at Maximum Augmentation
Fn_{sls}	Net Thrust at Sea Level Static and Maximum Augmentation
Fn/W	Aircraft Thrust Loading, Fn_{sls}/TOGW
Fn/Wgt	Engine Thrust to Weight Ratio, Fn_{sls}/Wgt_{eng}
FT	Feet
g	Constant 32.174 ft/sec^2
G's	Acceleration Normalized by Acceleration of Gravity
Knots	Nautical Miles per hour
$\dot{M}_9$	Mass Flow at Nozzle Exit Plane
ΔN	Changes in Normal Load Factor
N	Normal Load Factor
Ps	Specific Excess Power
psf	Pounds per Square Foot
P_8	Total Pressure at Nozzle Throat
P_{so}	Static Pressure-Ambient
P_{s9}	Static Pressure-Nozzle Exit Plane

R Aircraft Turn Radius-Instantaneous

S, S_w Wing Area

TOGW Takeoff Gross Weight of Aircraft ·

V Velocity-Freestream

V_9 Velocity at Nozzle Exit Plane

W Combat Weight of Aircraft (Except in Fn/W)

W_{fuel} Fuel Weight-Internal

W_{cool} Cooling Flow Rate

Wgt_{eng} Engine Weight Including Nozzle

$Wgt_{eng\&noz}$ Engine Weight Including Nozzle

X/L Nondimensional distance in freestream direction

δ_v, δ_{ap} Angle at Which Vectored Thrust Force is Applied, Measured Clockwise From Fuselage Reference Line (FRL).

ϕ Angle to Which Nozzle Exit Plane is Rotated Measured Clockwise From Line Normal to FRL

Δ Change in Value

2-D Two-Dimensional

C-D Convergent-Divergent

V-N Normal Load Factor Versus Airspeed Diagram

INTRODUCTION

Advanced nozzle technology is an extremely broad topic. Rather than trying to give a review of all the work in each of the technical subareas related to this broad topic, this paper concentrates on nonaxisymmetric configurations and the work being done in the technical subareas most critical to these types of configurations. It is important to bear in mind that neither the areas discussed nor the depth in any one area is all encompassing. However, the advanced techniques and technologies discussed are frequently equally applicable to conventional axisymmetric nozzle configurations so that the non-specialist in any one area can read the work as an overview of the entire area. It seems only fair to define the point of reference from which the paper was assembled. The authors view the nozzle as first and foremost a basic component of the engine and attempt to maintain this point of reference throughout. However, the importance of the interactions between the engine with its nozzle and the airframe is fully appreciated.

The following topics are discussed in the paper: The potential benefits which might be obtained through the use of nonaxisymmetric nozzles; a description of three nonaxisymmetric nozzle configurations; examples of the results of recent tests and analysis in the technical areas most critical to achieving acceptable engine performance with a non-axisymmetric nozzle; the level of installation benfits which must be obtained to make the installed performance comparable with what can be obtained with an axisymmetric nozzle; the use of thrust vectoring to enhance aircraft maneuverability and the importance of basic aircraft design parameters to the degree of enhancement which can be achieved; force accounting problems unique to the nonaxisymmetric or thrust vectoring nozzles; and, a brief discussion of the further development work which is required to bring these types of nozzles to the stage of development where they might be considered design options of acceptable risk.

POTENTIAL BENEFITS

There are a number of benefits which might be derived through the use of nonaxisymmetric nozzles. The most basic of these is improved integration of engine and airframe to obtain a better aerodynamic design. Benefits in the areas of detectability and therefore vulnerability also have been postulated based on directionalization and reduction of infrared signatures and radar cross sections. If the penalties for the use of these types of nozzles are offset by other benefits, then there can also be a potential reduced weight penalty for incorporation of either thrust reversal or thrust vectoring capabilities.

For the variety of reasons including twin engine reliability, improved aerodynamics at high angles of attack, and numerous other structural and aerodynamic consideratons, designers of combat aircraft have nearly abandoned their predispostion toward fuselages of circular cross section. Even our most recent single engine fighter aircraft, the F16, has a fuselage which upon close inspection is far from circular. The propulsion side of the industry has, however, steadfastly refused to consider turbines of other than circular cross sections, a trend which is likely to continue.

Blending of the airframe and engine to obtain superior aerodynamics can be one of the most difficult problems of aircraft design. Much of the zero lift drag of an aircraft occurs on the afterbody and nozzle and it is in this region that care must be taken to ensure good recompression and prevent separation in the resulting adverse pressure gradient. The complex three-dimensional contours required to blend axisymmetric nozzles with non-axisymmetric airframes complicate the problem by steepening the pressure gradients locally. The problem is further complicated by the need to keep nozzle length to a minimum for weight considerations and the need to incorporate variable exit area to obtain acceptable performance. Planar or nonaxisymmetric nozzles provide an excellent opportunity to reduce three-dimensional effects by spreading the recompression more uniformly around the cross section. Shown in Figure 1 is the aesthetically pleasing result of incorporating non-axisymmetric nozzles on a current air superiority aircraft. Unfortunately, quantitative assessment of the benefits actually derived on such a vehicle must await completion of extensive testing of such realistic configurations. So far, extensive drag data on non-axisymmetric nozzles has been limited to some basic research configurations. The USAF and NASA are currently conducting wind tunnel tests on more realistic fighter type configurations, and quantitative results should be available shortly.

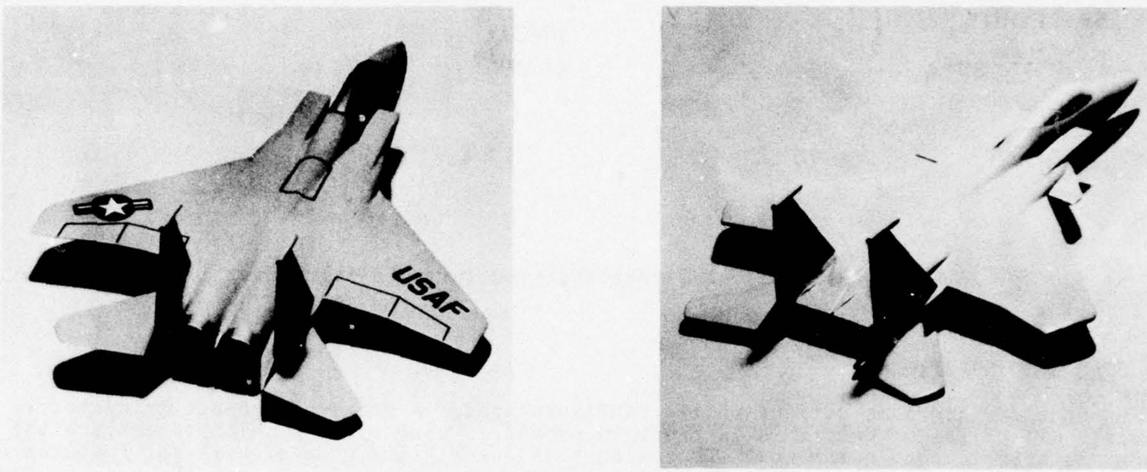


Fig.1 Air Superority Fighter with Axisymmetric and Nonaxisymmetric Nozzles

Another potential benefit of nonaxisymmetric nozzles is reduced and directionalized infrared signatures. The infrared signature consists of the radiation emanating from both the hot metal parts of the engine and nozzle and the hot exhaust gases. The use of non-axisymmetric nozzles can affect both of these sources of infrared radiation. The signature radiating from hot engine or nozzle parts can be reduced by providing line of sight blockage. This signature can be directionalized in cases where it cannot be reduced. On the left side of Figure 2 is a depiction of the intermediate power hot parts infrared signature of an engine with a conventional axisymmetric nozzle. By replacing the axisymmetric nozzle with a nonaxisymmetric plug nozzle which provides line of sight blockage, this signature can be reduced to the small ellipse depicted on the same figure. The right side of Figure 2 depicts how replacing an axisymmetric convergent-divergent nozzle with a single ramp, two-dimensional nozzle affects the directionalization of the infrared signature due to both hot parts and exhaust plume. This reduction in radiation emanating from the hot gases in the exhaust plume results from the increased mixing of the exhaust plume with the cooler external air of the freestream. The nonaxisymmetric concepts provide more circumference surrounding a given cross-sectional area of exhaust plume, thus providing more surface area for mixing. In some cases, the mixing is further enhanced by the formation of strong vortices in the vicinity of sidewalls and corners caused by the locally severe pressure gradients around these corners.

Nonaxisymmetric nozzles also offer the potential to reduce and directionalize radar cross section. The exhaust system is a major contributor to the radar cross section of a fighter aircraft when viewed from the rear. The primary factors affecting the radar return are the dimensions of the cavity formed by the engine, augmentor, and nozzle, and the shape and dimensions of the entrance to this cavity. While the types of nonaxisymmetric nozzles considered here do little to change the dimensions of the augmentor cavity, they do provide some degree of control over both the size and shape of the nozzle throat area which forms the entrance to the cavity. The variation in size is not in terms of flow area, but rather in terms of blockage of a line of sight into the cavity. In addition to providing line of sight blockage, nonaxisymmetric nozzles provide a surface for the application of radar absorbing materials to minimize reflection entering and exiting the cavity.

Nonaxisymmetric nozzles also have a potential advantage in terms of ease of incorporation of thrust reversal and thrust vectoring. Schemes to incorporate thrust vectoring and thrust reversal on axisymmetric nozzles frequently add significant complexity to the nozzle. In contrast, incorporation of thrust vectoring on some two-dimensional, nonaxisymmetric nozzles requires little more than adding the appropriate control functions to differen-

tially operate existing actuators and providing sufficient structure to carry and transmit the increased normal loads. Similarly, although not to the extent encountered with vectoring, the incorporation of thrust reversal capability into nonaxisymmetric nozzles is in some cases easier than incorporating the same capability into axisymmetric nozzles.

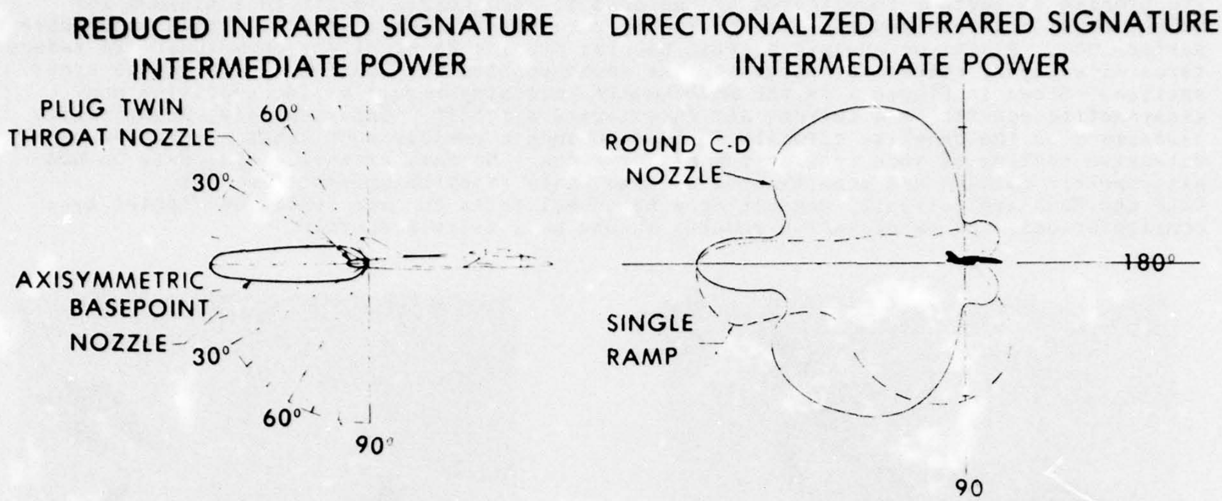


Fig.2 Reduced and Directionalized Infrared Signature

NOZZLE CONFIGURATIONS

Consider now some various nozzle configurations. A state-of-the-art axisymmetric nozzle will first be described in order to provide a base against which to compare the nonaxisymmetric configurations. Following this description, the significant features of each of three classes of nonaxisymmetric nozzles are described. All of the nozzles considered here are self-cooled by fan discharge air routed behind the augmentor liner. All of the configurations provide throat area variation suitable for augmented operation on turbofan cycles. Also, all designs feature exit area variation which may be either slaved to throat area or may be independently variable. All thrust vectoring is in the pitch plane only.

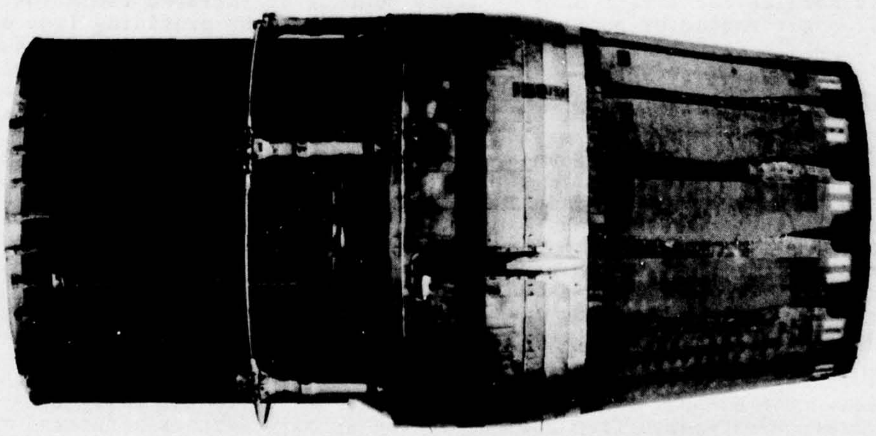


Fig.3 State-of-the-Art Axisymmetric Nozzle

An axisymmetric nozzle which is representative of the technology level found on most advanced current production fighter engines is shown in Figure 3. It has a peak gross thrust coefficient of over .985, and requires approximately 8% of the engine airflow for cooling. Throat area variation is accomplished with overlapping flaps and seals. Thrust reversing capability can be incorporated by the use of a blocker and cascade arrangement, and thrust vectoring can be achieved through use of a dual gimbal as shown in Figure 4.

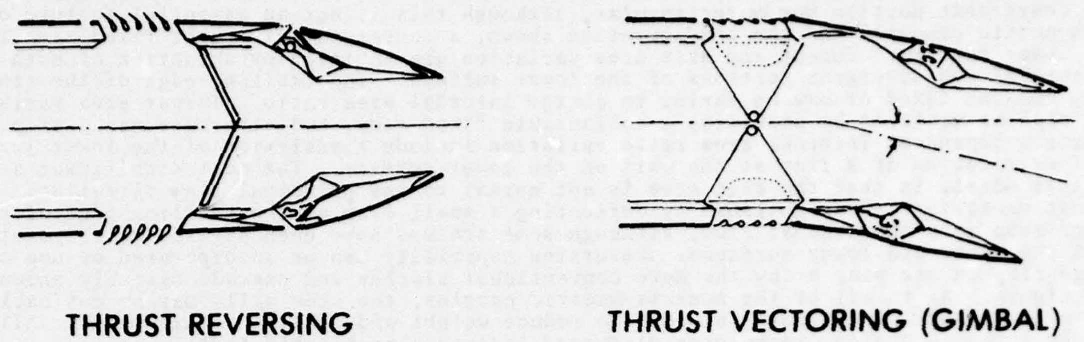


Fig.4 Reversing and Vectoring Axisymmetric Nozzle

In Figure 5 a simple two-dimensional, convergent-divergent nozzle is depicted. The cross sections at the exit, the throat, and the inlet to the convergent portion of the nozzle are all rectangular. As with all nonaxisymmetric configurations, there is a duct upstream which provides the transition from the circular cross section of the engine. The convergent section of the nozzle is formed by two planar flaps, and throat area variation is achieved by actuators moving these two surfaces. A second pair of planar flaps forms the divergent section. The exit area may be slaved to the throat area to provide a specified area ratio schedule, or a second set of actuators can be used to provide independent exit area variation. If independent area ratio variation is incorporated, thrust vectoring can be achieved by differential use of the same actuators. Thrust reversal can be accomplished by arranging so that rotation of the convergent flaps past the minimum throat area position opens passages on the upper and lower surfaces of the nozzle. As the convergent flaps are rotated further, the total flow area transitions to these upper and lower openings. Several cooling techniques are applicable to this type of nozzle. Film cooling, film plus convection cooling, fin wall, as well as some more advanced techniques could be used effectively. These cooling techniques will be discussed in more detail later.

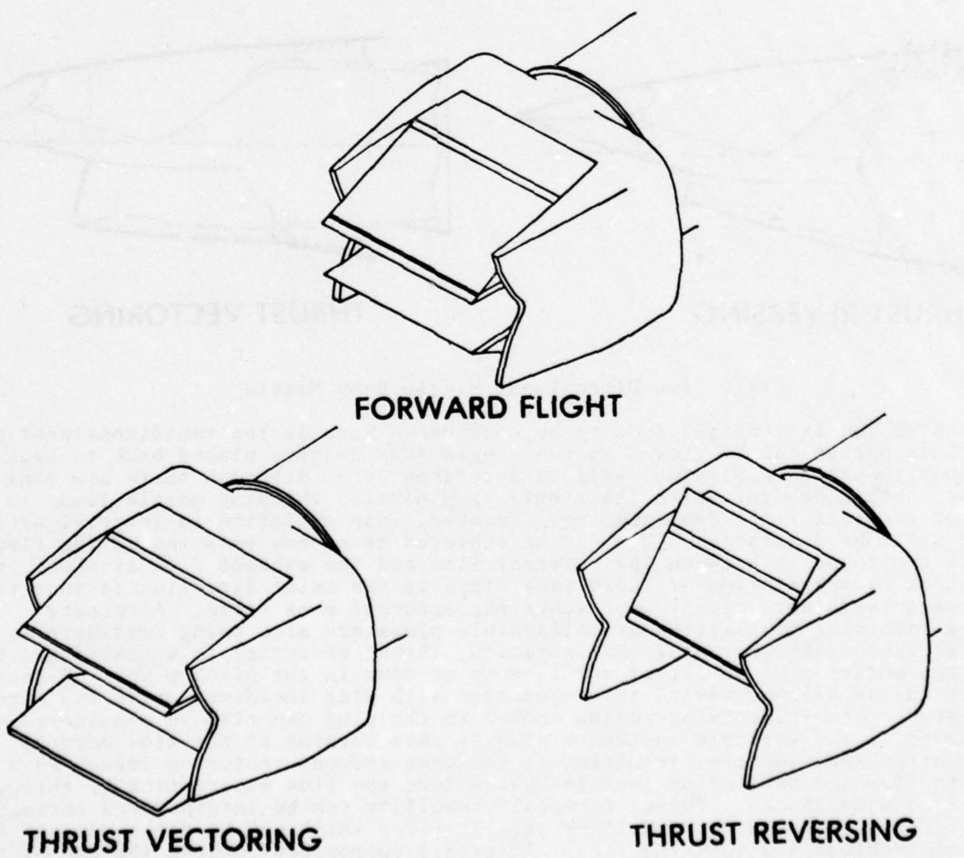
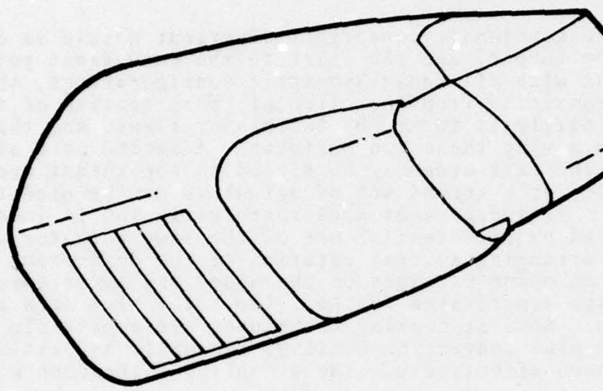


Fig.5 Two-Dimensional, Convergent-Divergent Nozzle

Another type of nonaxisymmetric nozzle is a two-dimensional, single ramp nozzle shown in Figure 6. Again the cross-sections at the exit, the throat, and the inlet to the convergent portion may be rectangular, although this is not an essential feature of this nozzle concept. In the configuration shown, a convergent-divergent fixed ramp forms the upper surface. Throat and exit area variation are achieved by actuation of both the convergent and divergent portions of the lower surface. The trailing edge of the single ramp remains fixed or may be varied to change internal area ratio. Throat area variation can also be achieved by providing a collapsible fixed ramp, and alternate means of providing independent internal area ratio variation include translation of the lower surface cowl or rotation of a flap at the exit on the lower surface. The most significant aspect of this nozzle is that the exit area is not normal to the principal flow direction. Thrust vectoring is accomplished by deflecting a small flap at the trailing edge of the upper ramp into the exhaust flow, although some schemes have been devised which rotate both the upper and lower surfaces. Reversing capability can be incorporated by use of a large flap on the plug or by the more conventional blocker and cascade assembly shown in the figure. As in all of the nonaxisymmetric nozzles, the side walls may be cut back in any of a variety of contours in order to reduce weight and cooling requirements. All of the advanced cooling techniques discussed later are applicable to this concept.



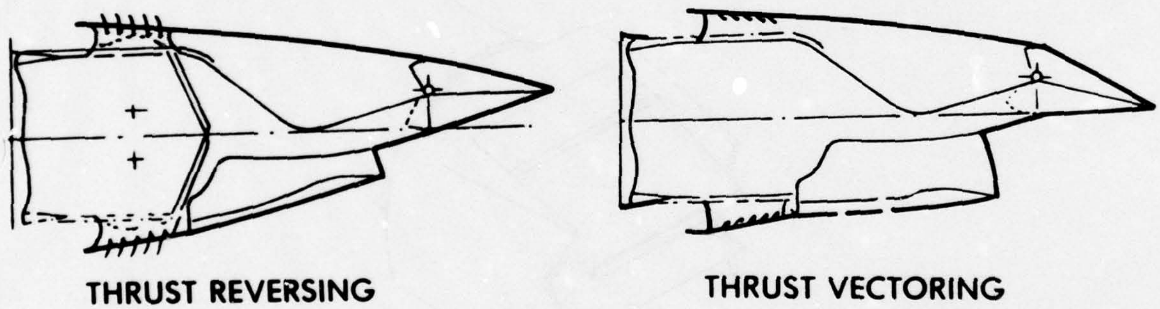


Fig.6 Two-Dimensional Single Ramp Nozzle

The third and last nozzle type to be considered here is the two-dimensional plug nozzle. This nozzle can be viewed as two single ramp nozzles placed back to back. The plug nozzle shown in Figure 7 will be described here, although there are many variations on this design. Like the single ramp nozzle, the plug nozzle tends to be altitude or pressure ratio compensating. However, some variation in internal area ratio may still be desirable. This can be achieved by either rotation of the flaps which form the boundary between the external flow and the exhaust flow as shown in the figure or by translation of these same flaps in the axial direction so that the internal area ratio more nearly approaches the external area ratio. Alternate approaches involving translating or collapsible plugs are also being considered. With the variable incidence plug configuration, thrust vectoring is accomplished by pivoting the entire plug to direct the flow up or down in the pitch plane. Divergent flap position can also be varied in conjunction with plug incidence angle and more sophisticated schemes involving adding camber to the plug can also be considered. One advantage of the variable incidence plug is that turning of the flow occurs primarily at subsonic speeds, resulting in somewhat reduced vectoring losses when compared to flap and deflection surfaces which turn the flow supersonically through the use of oblique shocks. Thrust reversal capability can be incorporated through the use of flaps on the upper and lower plug surfaces which may be deployed into the exhaust jet to block and turn the flow. Alternate approaches include the use of a more conventional blocker and cascade upstream of the plug, or using the convergent flaps in a manner similar to that described for the two-dimensional convergent-divergent nozzle. Advanced cooling techniques are required for the two-dimensional plug nozzle because of

the substantially increased surface areawetted by the exhaust gases.

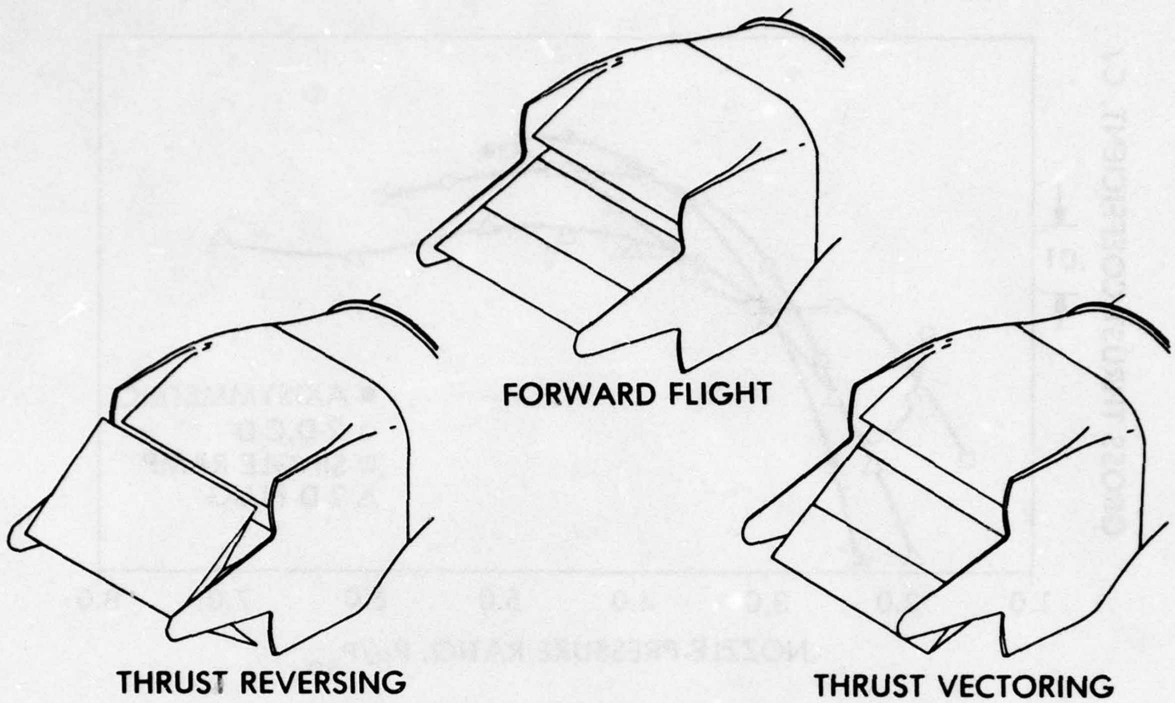


Fig.7 Two-Dimensional Plug Nozzle

CRITICAL TECHNOLOGIES

Before nonaxisymmetric nozzles can be seriously considered as an option available to the designer, it is necessary to demonstrate that the level of performance achievable with these nozzles is at least comparable to that achievable with an axisymmetric concept. Such a demonstration requires development and demonstration of the level of performance achievable in certain critical technical areas: internal performance, cooling requirements, and weight or structural integrity.

The primary index of nozzle internal performance is the gross thrust coefficient. The importance of attaining high levels of gross thrust coefficient is self evident, since this acts as a direct multiplier on the total gross thrust produced. It is important for a nonaxisymmetric nozzle to have a high gross thrust coefficient at the design point. For a given flap length and maximum divergence angle, the nonaxisymmetric nozzle provides less area ratio variation capability than does an axisymmetric nozzle. Figure 8 depicts the variation of gross thrust with nozzle pressure ratio for each of these advanced nozzle concepts considered, as well as the axisymmetric configuration. This data was obtained from recent subscale static tests. Note that the peak gross thrust coefficient when compared to an axisymmetric nozzle is decreased by one half percent for the two-dimensional, convergent-divergent nozzle, .25 percent for the single ramp nozzle, and 1.25 percent for the 2-D plug nozzle. The off-design performance also is depicted on this figure. Thrust vectoring also affects the gross thrust coefficient, and the variation in gross thrust coefficient which results as the vector angle is changed can be seen in Figure 9. Some caution must be used in interpreting or using this data because the traditional definition of gross thrust coefficient used here is not well suited to certain types of nonaxisymmetric nozzles at off-design conditions or in the vectored mode. This will be discussed in more detail later. Also, it should be emphasized that these results are for static performance only and may be altered for the configurations where external flow can affect the exhaust flow field thus resulting in changes in the pressure forces applied to the nozzle surfaces.

Nonaxisymmetric nozzles usually present more of a cooling problem than axisymmetric nozzles. The increased surface area wetted by the hot exhaust gas, and presence of corners and other high heat flux regions increase the cooling requirements for these types of nozzles. This need for increased cooling can be reduced somewhat through the judicious use of techniques such as sidewall cutback, but this is normally detrimental to internal performance. The cooling flow required principally affects the maximum augmented thrust available, since air which is required for cooling is not available for augmentation.

There are a number of advanced cooling techniques under consideration which may be used to achieve higher cooling effectiveness and therefore reduce the required cooling flow. Three cooling techniques are depicted in Figure 10. The first and most common technique in current use is film cooling. In this method, a thin film of cooler gas is used to protect the surface from the hot exhaust gas. A second method is multiple panel,

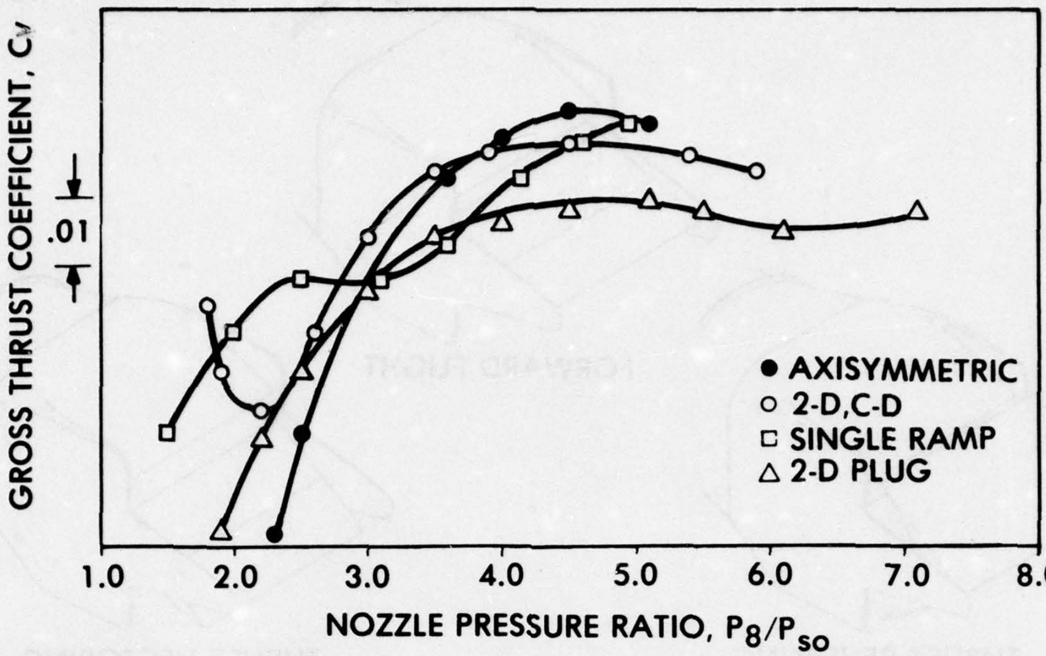


Fig.8 Nozzle Internal Performance

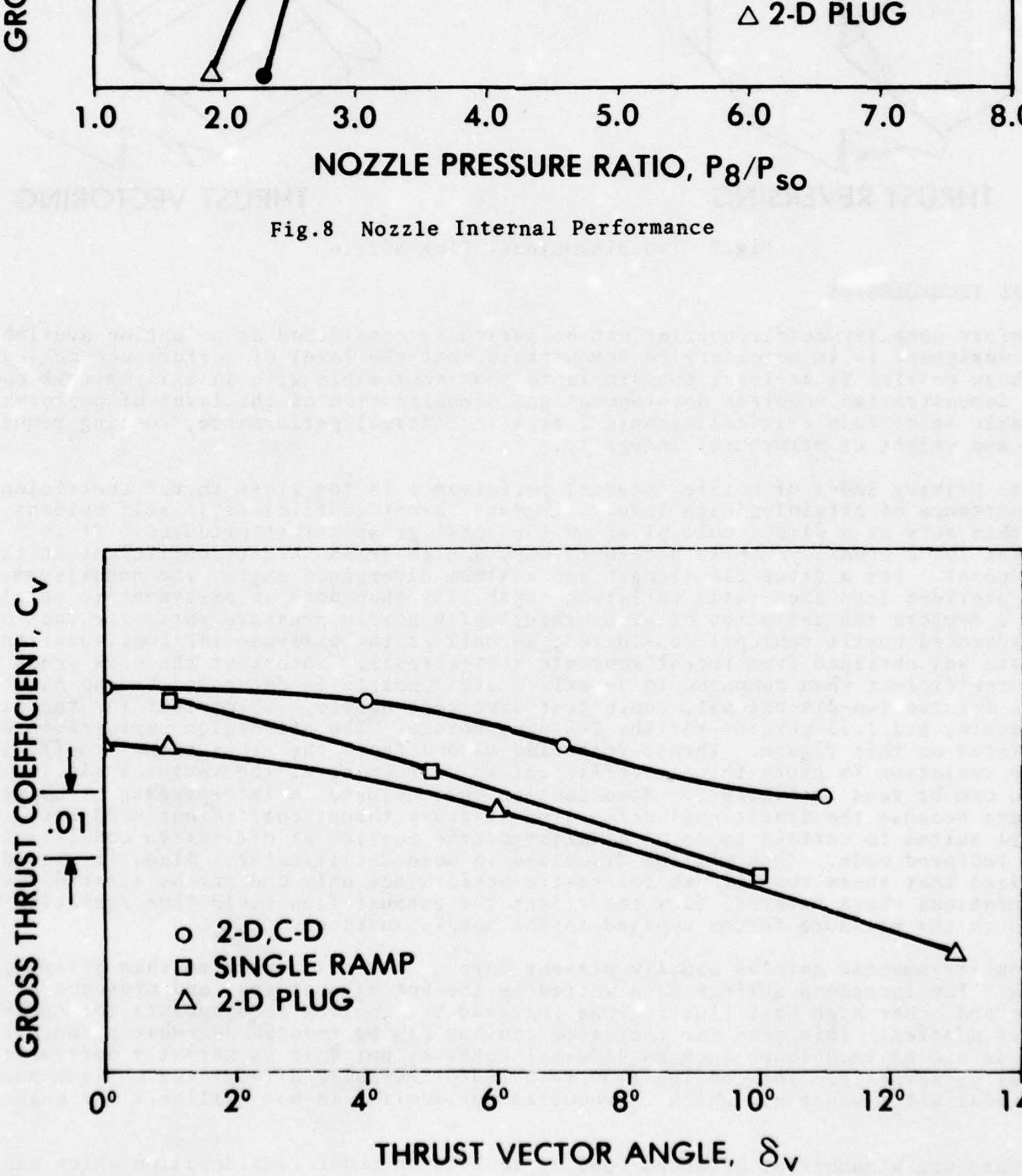


Fig.9 Effect of Vectoring on Internal Performance

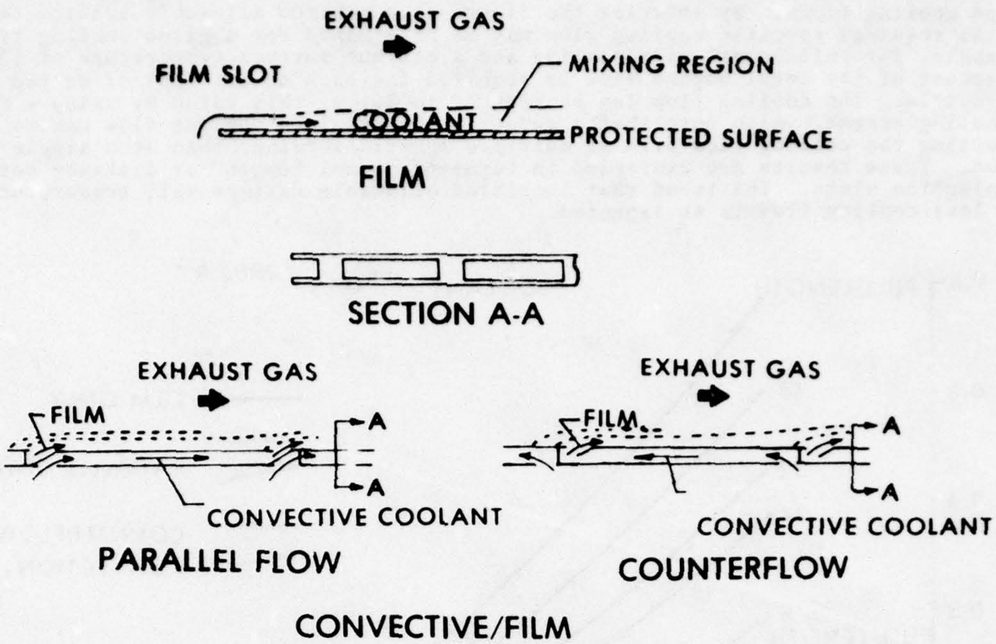


Fig.10 Nozzle Cooling Techniques

parallel flow finwall. This is really a combination of film and convective cooling methods. The cooling air flows from a reservoir through a channel running parallel to the mainstream and provides convective cooling to the channel surfaces. Upon reaching the end of this channel the cooling flow is discharged to form a film, thus providing film cooling for the surface of the succeeding panel. A natural extension of this principal is the counter flow finwall method. By reversing the panels, the flow in the convective passages may be made to flow counter to the mainstream flow, thus reducing the axial temperature gradients along and through the wall. This in turn reduces the buckling and bending tendency of the panels. Figure 11 shows an example of the hot wall surface temperature distribution in the main stream flow direction for both parallel flow and counterflow finwall. Note that the counterflow finwall technique results in a more uniform temperature distribution along the panel. This allows the required cooling flow to be reduced to half of that which would be required using parallel flow finwall, and also provides improvements in the structural area by reducing the temperature gradient along the hot wall.

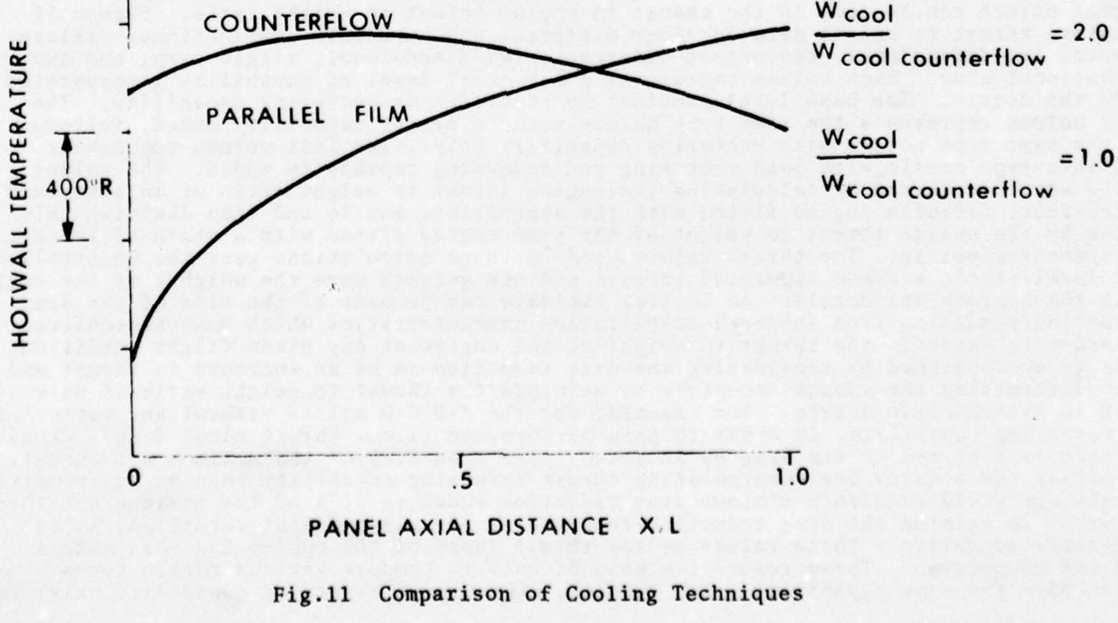


Fig.11 Comparison of Cooling Techniques

Figure 12 provides a summary of the benefits to be gained through the use of advanced cooling flows. By entering the figure at a maximum allowable surface temperature, the required specific cooling flow may be determined for a given cooling type. For example, for full length film cooling and a maximum surface temperature of 1400°F, 0.35 percent of the total engine flow is required for each square foot of wetted area in the nozzle. The cooling flow can be reduced to 25% of this value by using a counter flow cooling scheme. Also note that a reduction in required cooling flow can be achieved by injecting the cooling flow film at multiple locations rather than at a single upstream location. These results are expressed in terms of "panel length" or distance between film injection slots. The trend that increased allowable maximum wall temperature requires less cooling flow is as expected.

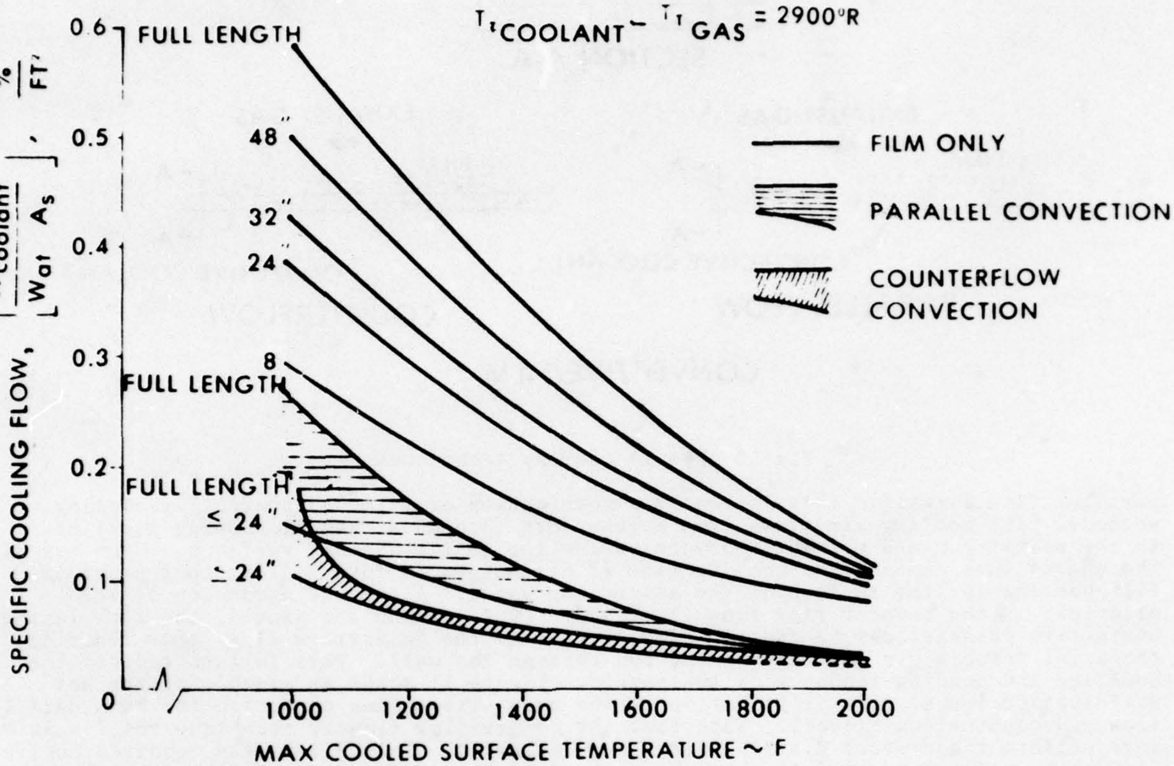


Fig.12 Effect of Nozzle Cooling Techniques

Thus far a number of potential benefits which might be derived through the use of nonaxisymmetric nozzles have been considered, along with some potential nozzle designs. It is appropriate now to consider the penalties in performance against which these potential benefits must be balanced. Although nonaxisymmetric nozzles will influence the specific fuel consumption through changes in the gross thrust coefficient, a more fundamental effect can be seen in the change in engine thrust to weight ratio. Figure 13 displays thrust to weight data for four different engine-nozzle combinations: axisymmetric; two-dimensional, convergent-divergent; two-dimensional, single ramp; and two-dimensional plug. Each column represents a different level of capability incorporated into the nozzle. The base level provides no reversing or vectoring capability. The next column represents the same type nozzle with reversing capability added, followed by the same type nozzle with vectoring capability only. The last column represents the same type nozzle with both vectoring and reversing capability added. The values shown were determined by calculating the engine thrust to weight ratio of an afterburning, mixed-flow, turbofan engine fitted with the appropriate nozzle and then dividing this value by the engine thrust to weight of the same engine fitted with a state-of-the-art axisymmetric nozzle. The thrust values used in these calculations were the uninstalled sea level static maximum augmented thrusts and the weights were the weights of the engine plus the appropriate nozzle. An initial estimate can be made of the size of the drag reduction resulting from improved installation characteristics which must be achieved in order to maintain the thrust to weight of the engine at any given flight condition. This is accomplished by considering the drag reduction to be an increase in thrust and then determining the amount necessary to maintain the thrust to weight ratio of an engine with an axisymmetric nozzle. For example, for the 2-D C-D nozzle without any vectoring or reversing capability, in order to gain performance (i.e., thrust minus drag), it will be necessary to reduce the drag by an amount more than 3.9% of the maximum net thrust. To offset the penalty for incorporating thrust reversing capability into an axisymmetric nozzle, it would require a minimum drag reduction equal to 5.7% of the maximum net thrust. In order to examine the drag reduction required at any given flight condition, it is necessary to multiply these values by the thrust lapse of the engine and then make a similar comparison. These values can also be used to compare various nozzle types which have the same capability. For example, with thrust reversing capability only, the

uninstalled performance of the nonaxisymmetric nozzle is competitive to marginally superior to that of the axisymmetric nozzle. For vectoring only, the nonaxisymmetric nozzles are less than competitive, and when both vectoring and reversing capability is required nonaxisymmetric nozzles are slightly superior.

ENGINE THRUST/WEIGHT RELATIVE TO AXISYMMETRIC

NOZZLE TYPE	BASE	REVERSING ONLY	VECTORING ONLY	VECTORING AND REVERSING
AXISYMMETRIC	1.000	.911-.943	.967-.974	.893-.914
2-D C-D AR=1.55	.961	.947	.944	.939
2-D SINGLE RAMP	.981	.934	.967	.921
2-D PLUG	.952	.911	.926	.896

SEA LEVEL STATIC

Fig.13 Integrated Effect On Engine Performance

Figure 14 takes the last column of Figure 13, the vectoring and reversing case, and shows the contributions made by cooling requirements, internal performance, and weight to the changes in thrust to weight ratio of the engines being considered. All data presented is for maximum augmented thrust at sea level static conditions. The thrust and thrust to weight ratio for each engine-nozzle combination was based on the same airflow size. Note that while the vectoring and reversal case is the only one considered in this figure, the top row of the matrix still depicts the axisymmetric case (without either vectoring or reversal capability) to which the other data is referenced. Note that the use of advanced cooling techniques can result in a decrease in the required cooling flow. This beneficial effect coupled with improved internal performance (at sea level static conditions) results in an increase in net thrust. The major driver on engine thrust to weight, however, is the increased weight of the nozzles. The overall result is that when thrust vectoring and reversing capabilities are required, a nonaxisymmetric nozzle weighs slightly less and produces more thrust than an axisymmetric nozzle with the same capability.

	NOZZLE TYPE	COOLING FLOW	Cv	Fn	Wgt noz+eng	Fn Wgt
W.O.	AXISYMMETRIC	1.000	1.000	1.000	1.000	1.000
W VECTORING AND REVERSAL	AXISYMMETRIC	1.000	1.000	1.000	1.09-1.12	.893-.914
	2D-CD, AR=1.55	.385	1.013	1.025	1.09	.939
	2D SINGLE RAMP	1.000	1.006	1.006	1.09	.921
	2D PLUG	.50	1.026	1.037	1.16	.896

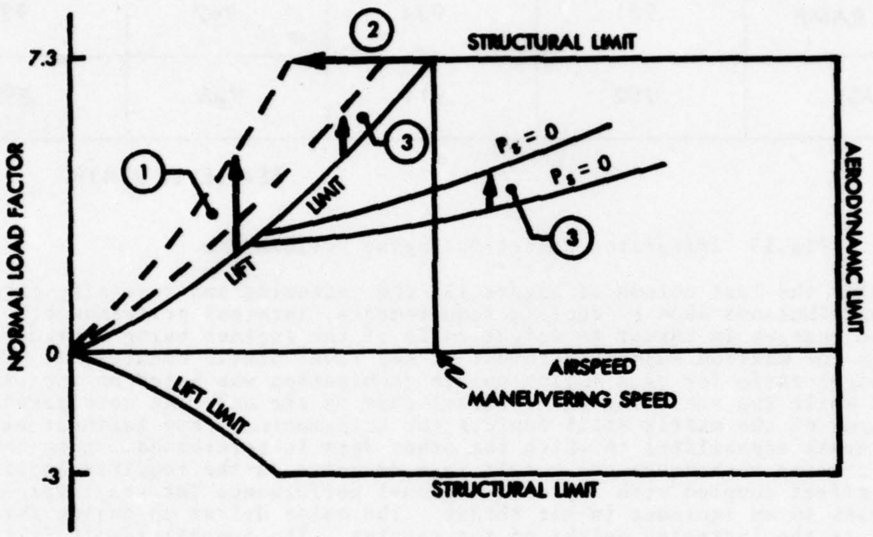
SEA LEVEL STATIC

Fig.14 Individual Contributions to Integrated Engine Performance

The previous discussion has shown that on the basis of uninstalled performance, nonaxisymmetric nozzles offer a clear advantage only when both thrust vectoring and reversing are required. In the other cases, some installation benefits must be obtained merely to offset the increased weight of the nonaxisymmetric nozzle. Since the question of whether nonaxisymmetric designs provide benefits when thrust vectoring is not required cannot be answered until quantitative installation data is available, it is appropriate to consider the question of whether thrust vectoring offers any potential advantages.

THRUST VECTORING ENHANCEMENT OF AIRCRAFT MANEUVER CAPABILITY

Since ease of incorporation of thrust vectoring capability is one of the potential benefits of nonaxisymmetric nozzles, it is logical to examine whether any significant benefit can be achieved through the use of thrust vectoring. It is important to note at this point that the question of the value of thrust vectoring is a separate question from the value of nonaxisymmetric nozzles. Too often these are confused as the same question. One must also remember that at the high angles of attack often encountered during maneuvering, today's aircraft experience a substantial force in the lift direction from engine thrust. So the question is not whether vectoring the thrust to achieve a force in the lift direction is beneficial, but whether substantial benefits can be obtained by a thrust vector schedule other than that where the vector angle equals the angle-of-attack. A study was conducted to examine this question, and to determine the influence of basic aircraft design variables such as thrust loading, Fn/TOGW, and wing loading, W/S, on the size of the improvements achievable through the use of thrust vectoring.



PAYOFFS EXAMINED:

1 INCREASE IN MAXIMUM NORMAL LOAD FACTOR (INSTANTANEOUS)

2 DECREASE IN MANEUVERING SPEED

3 NORMAL LOAD FACTOR INCREASE (INSTANTANEOUS AND SUSTAINED) WITHOUT A CORRESPONDING P_s PENALTY

Fig.15 Velocity-Normal Load Factor Diagram

The ability to thrust vector has a number of potential benefits, one of which is an increase in normal load factor. To understand this benefit, consider first a typical V-N diagram for a combat aircraft (Figure 15). The left-hand side of the envelope is bounded by lift limit curves (both positive and negative). This represents the maximum number of G's achievable at a given velocity at the maximum angle-of-attack. The top and bottom of the envelope are bounded by structural limits imposed by the air-frame. The right side of the envelope is bounded by aerodynamic limits imposed by such things stability and control problems, and stagnation temperature.

While an aircraft's G capability at a given velocity is only one of many capabilities relating to combat effectiveness, it is, nonetheless, a very important one. Velocity and normal load factor are related to turn radius by the equation:

$$R = \frac{V^2}{g(N^2 - 1)^{\frac{1}{2}}}$$

To minimize turn radius, it is desirable to minimize velocity while maximizing the normal load factor. The minimum turn radius usually occurs at a velocity slightly below that where the lift limit curve intersects the structural limit. For purposes of discussion, it will be assumed that the minimum turn radius occurs at that intersection (i.e., the minimum velocity at which the maximum allowable load factor can be obtained). This velocity will be referred to as "maneuvering speed" (see Figure 15).

Payoff can be obtained in a number of ways. It could be due to lowering the maneuvering speed, increasing the normal load factor obtainable below the maneuvering speed, or by decreasing the Ps penalty associated with a given maneuver. This effort explored these areas. A detailed analysis of the payoff in these areas was not intended, rather, a first approximation of the increase in maneuverability resulting from thrust vectoring was desired. The payoffs which were considered are depicted in Figure 15.

A number of assumptions were made in order to define the problem and limit the scope and complexity of the study. Careful consideration should be given to these assumptions when drawing conclusions from the results.

TABLE A

1. Aircraft Configuration	air superiority fighter
Combat Wing Loading (TOGW - .5 W_{fuel}/S_w)	55 lbs/ft²
Aircraft Thrust Loading $F_{n_{SLS}}$/TOGW	1.16
Internal Fuel Fraction W_{fuel}/TOGW	.30
Canard (only on thrust vectoring aircraft)	located forward of c.g., 20% of baseline wing area
2. Pitch Trim	
without vectoring	horizontal stabilizer only
with vectoring	
wing and fuselage	horizontal stabilizer only
vectored thrust	canard only
3. Aerodynamics	
Basic Polar	typical of air superiority fighter
Vectoring induced changes	none
Canard induced changes	add lift and drag of isolated canard-body (no interaction with wing)
4. Thrust Vectoring	vector gross thrust
C_v decrement	none
5. Weight	
Nozzle	no penalty
Canard	no penalty

The maneuver studied was a level turn at maximum power at 30,000 ft and combat weight, 50 percent internal fuel weight.

Both instantaneous normal load factor capability and maximum normal load factor capability were examined. The maximum instantaneous G limit was defined by the maximum trimmed angle-of-attack since no maximum was reached in the lift curve below 40 degrees angle-of-attack. No dynamic overshoot was included. Maximum sustained G capability was defined as the point where specific excess power equals zero.

EFFECT OF THRUST VECTORING ON NORMAL LOAD FACTOR

The first payoff considered was the increase in maximum number of G's at maximum angle-of-attack. The lift limit curve was first defined for the baseline aircraft. Next, various amounts of thrust vectoring were added (from 15 to 60° of vectoring). The results are depicted in Figure 16. Since the number of normal G's is directly proportional to the sum of the forces in the lift direction, it is logical that if one is interested only in maximizing the normal load factor, the maximum amount of thrust vectoring is desired. However, it is important to note that in most cases, the maximum number of G's is attainable only by greatly penalizing the specific excess power, P_s.

$$P_s = \frac{Fg \cos(\alpha + \delta_v) - (D_{ram} + D_{a/c} + D_{can}) V}{\text{Weight}}$$

and is also equal to aircraft's instantaneous rate of climb. In Figure 17, the values of specific excess power corresponding to the lift limit curves in Figure 16 are depicted.

(In both Figures 16 and 17, each curve has a left hand limit, which corresponds to the point where the maximum angle-of-attack is reached on the canard. Since any further decrease in Mach number would result in a situation where the pitching moment induced by thrust vectoring could not be trimmed out, the only recourse is to decrease the thrust vector angle.) Note the magnitude of the Ps penalty. At Mach .75, Ps ranges from approximately -850 feet per second with no thrust vectoring to -1800 feet per second with 60 degrees of thrust vectoring. This would result in an increase in the deceleration along the flight path from 1.1 to 2.4 G's. Therefore, the aircraft must pay a very large penalty in energy and must either decelerate, descend, or both in order to maximize its load factor. This is not to say that there are not occasions where a pilot would be willing to sacrifice energy in order to achieve a tight turn. It is only important to point out that while thrust vectoring can increase maximum load factor, there is a penalty tied to that payoff.

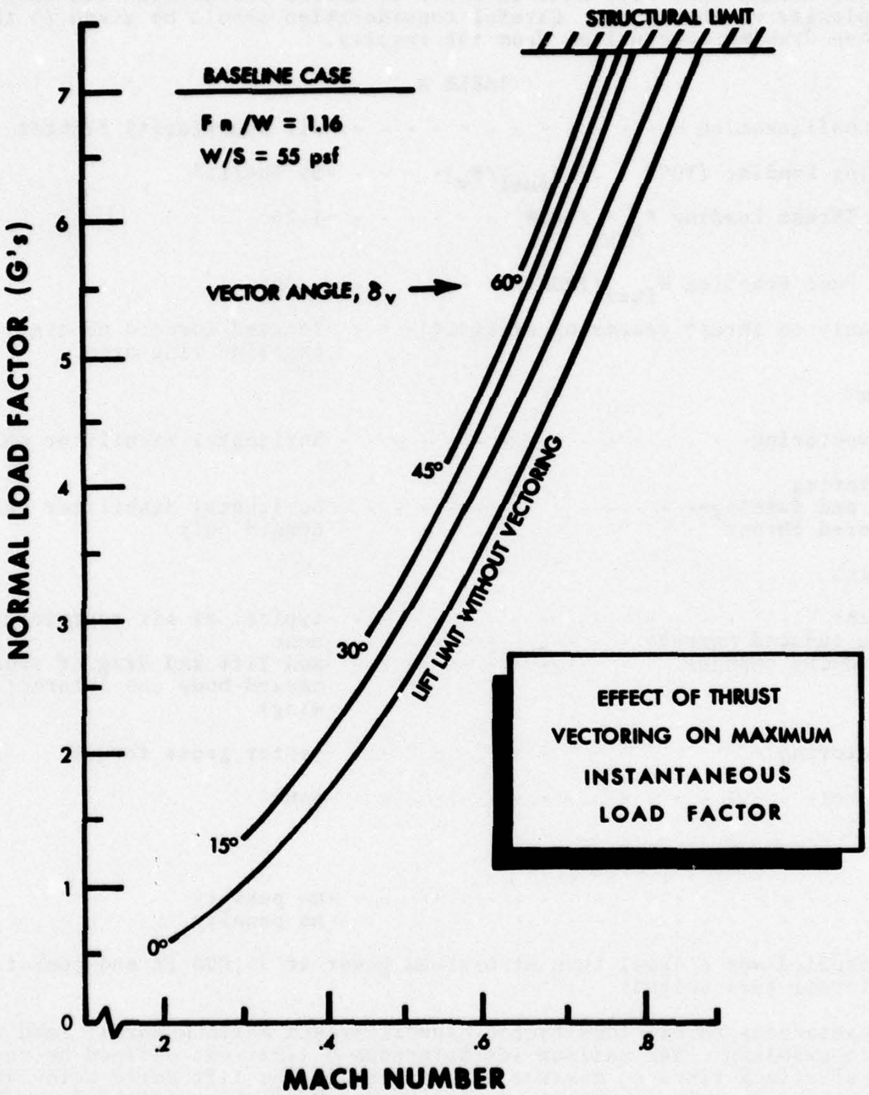


Fig.16 Effect on Instantaneous Normal Load Factor

Note from the previous discussion that at an altitude of 30,000 feet the aircraft cannot achieve maximum normal load factor at maneuvering speed without suffering a Ps penalty. However if the load factor corresponding to the structural limit can be achieved at a lower airspeed, the result will be a smaller turn radius. For example, if the aircraft were at the maximum structural limit of 7.3 G's, going from zero thrust vectoring to 15° of thrust vectoring would decrease maneuvering speed by approximately .04 Mach or about 25 knots (see Figure 16). Each successive increment of 15° additional vectoring capability results in successively smaller decreases in maneuvering speed. Figure 18 depicts the effect of vector angle on maneuvering speed and instantaneous turn radius. This payoff is again at the expense of Ps, as depicted in Figure 17.

So far each normal load factor payoff considered has had an associated Ps penalty. It is quite difficult to judge the value of a payoff coupled with a corresponding penalty without delving into the realm of combat tactics, which is beyond the scope of this

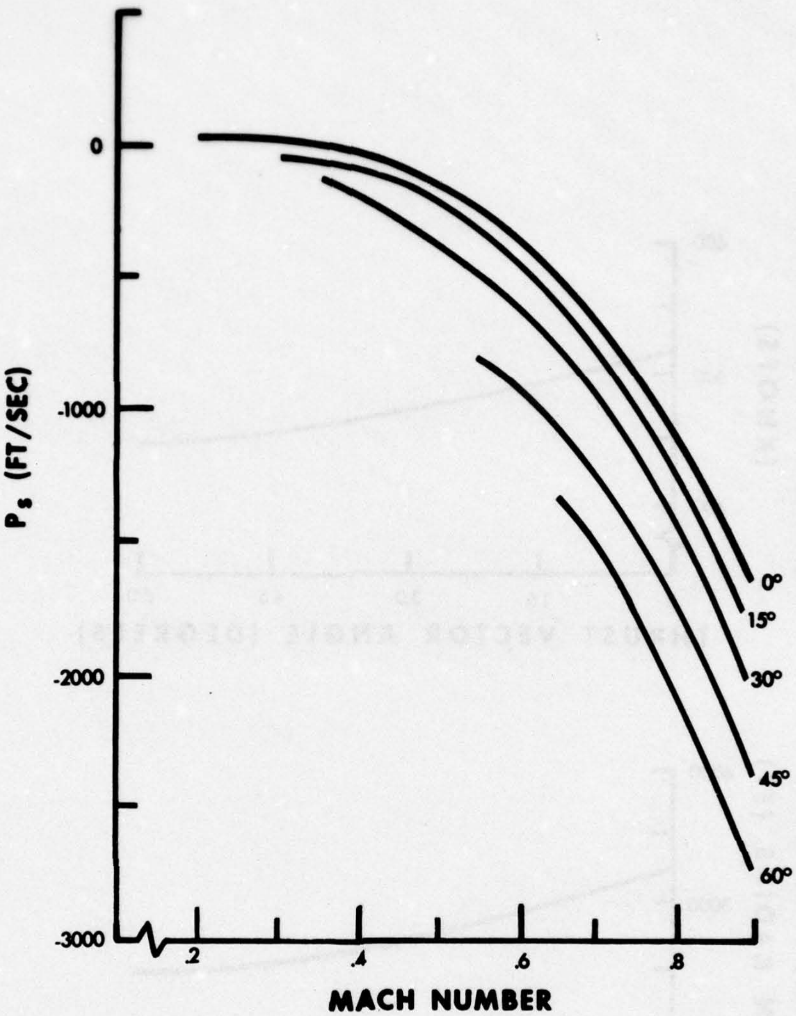


Fig.17 Effect on Specific Excess Power

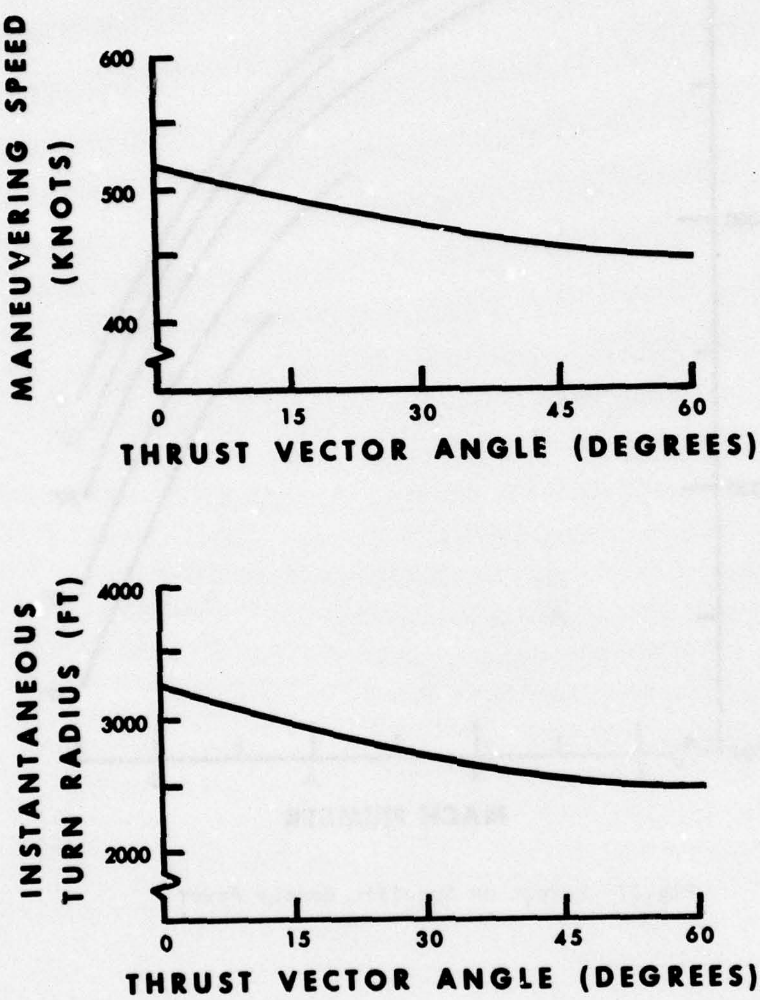


Fig.18 Effect on Maneuver Speed and Turn Radius

effort. However, it is logical to ask whether there is any normal load factor payoff which does not have a corresponding Ps penalty. An attempt to answer that question was made. Two important limits in the maneuvering envelope were considered: the maximum instantaneous G limit and the maximum sustained G limit. By adding thrust vectoring capability and the canard to maintain trim, it was possible to decrease an aircraft's angle-of-attack while maintaining the same G capability and gaining specific excess power. This gain in specific excess power could then be traded away for additional normal load factor capability. This was done for various thrust vector angles to determine an optimum thrust vector schedule and the resultant increase in normal load factor capability. Curve 2 in Figure 19 depicts the maximum instantaneous G limit for the baseline aircraft without vectoring. Curve 1 depicts the instantaneous G limit after thrust vectoring capability is added, while maintaining the Ps level achieved at the same velocity on Curve 2.

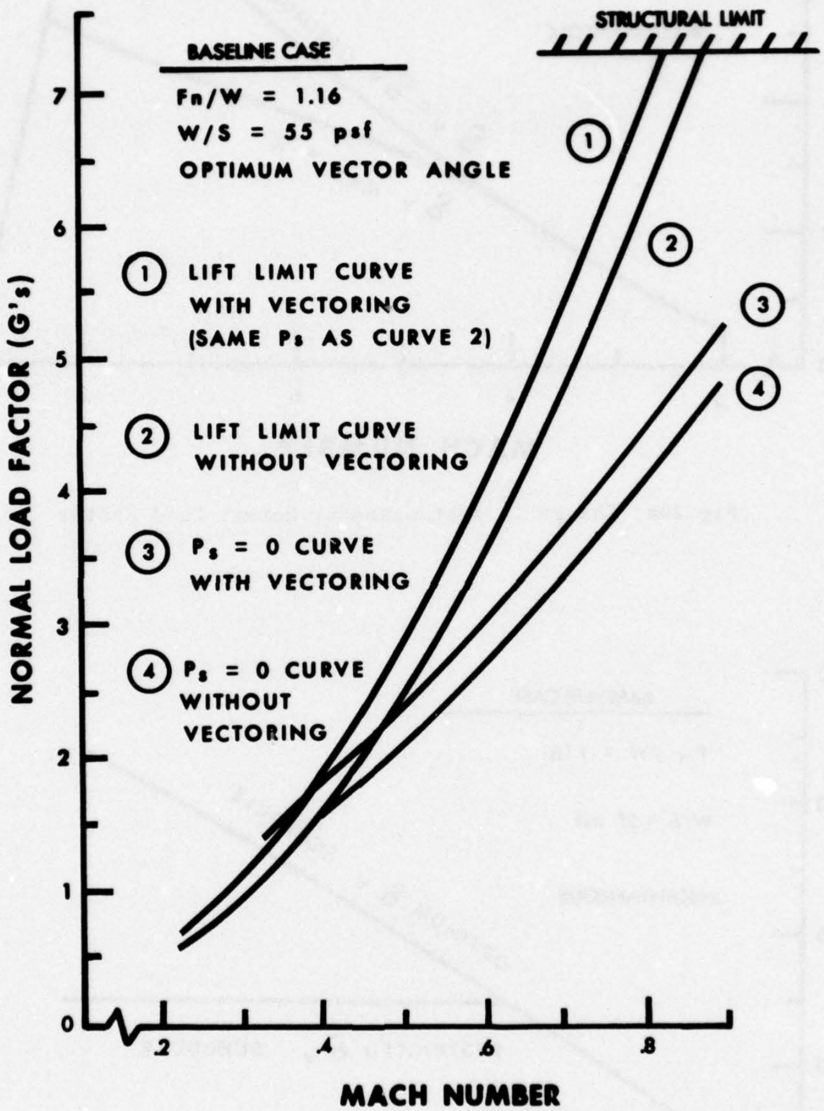


Fig.19 V-N Diagram, Baseline Case

Next, the sustained G limit curve for the nonvectored aircraft was calculated (Curve 4 on Figure 19). Thrust vectoring capability was then added, and a new Ps = 0 curve was obtained, again using the angle-of-attack and thrust vector angle at each velocity which resulted in the highest normal load factor (Curve 3). In order to better depict the payoff due to vectoring, the difference between curves 1 and 2 and between curves 3 and 4 has been plotted in Figures 20 and 21 respectively, along with the corresponding thrust vector schedules. Note that any time Δ normal load factor is plotted, it represents the difference between the G capability of an aircraft with thrust vectoring and the associated canard, and an identical aircraft without thrust vectoring or canard. By comparing the Δ normal load factors for aircraft with different thrust loadings and wing loadings, it is possible to determine what type of aircraft would benefit most by adding thrust vectoring capability.

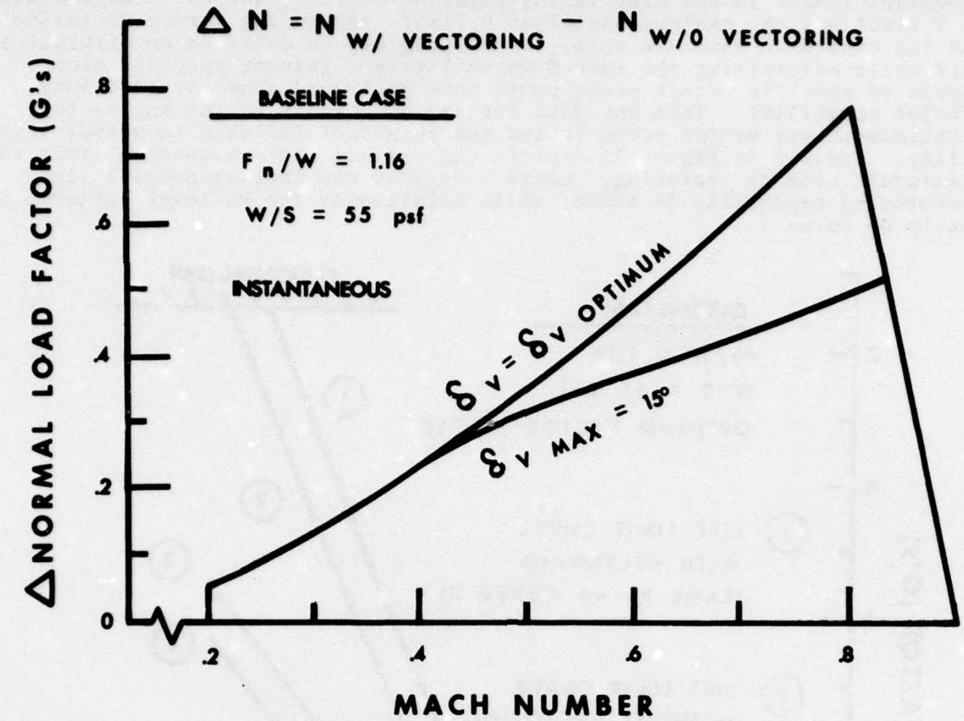


Fig.20a Change In Instantaneous Normal Load Factor

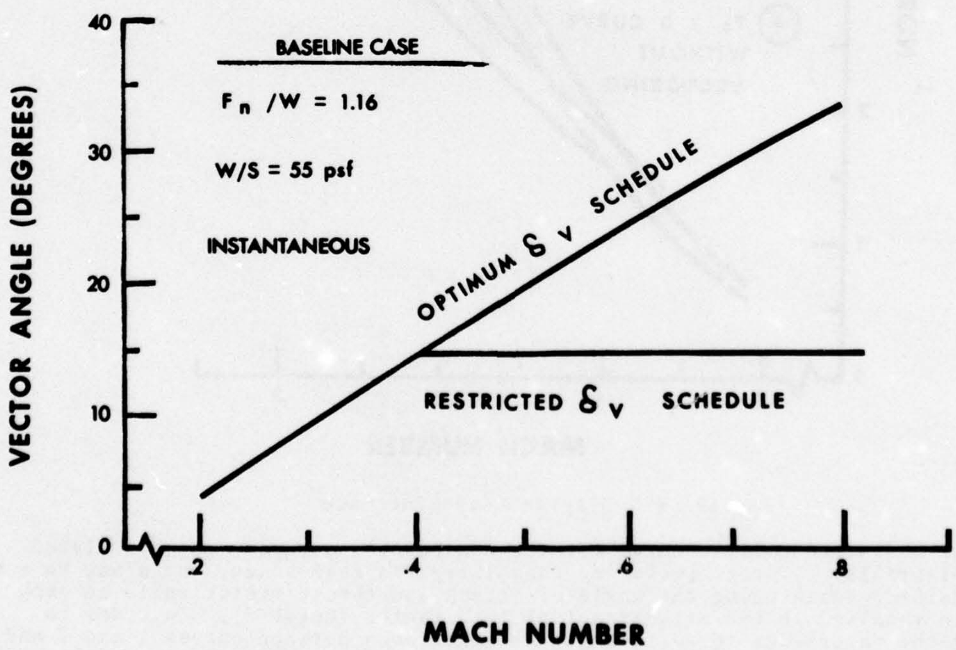


Fig.20b Thrust Vector Schedule - Instantaneous Turn

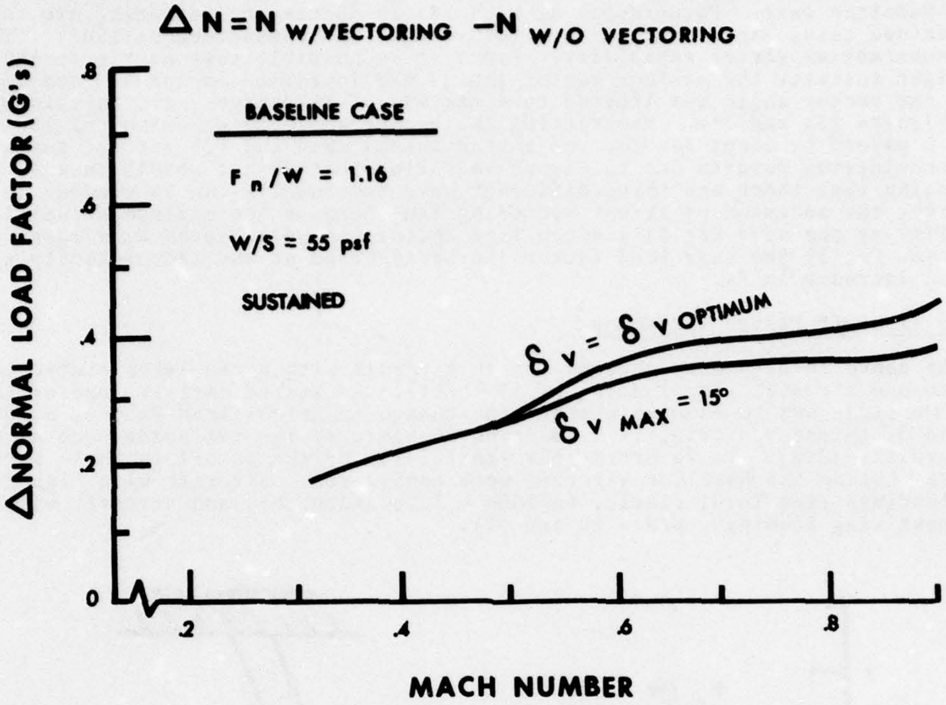


Fig.21a Change In Sustained Normal Load Factor

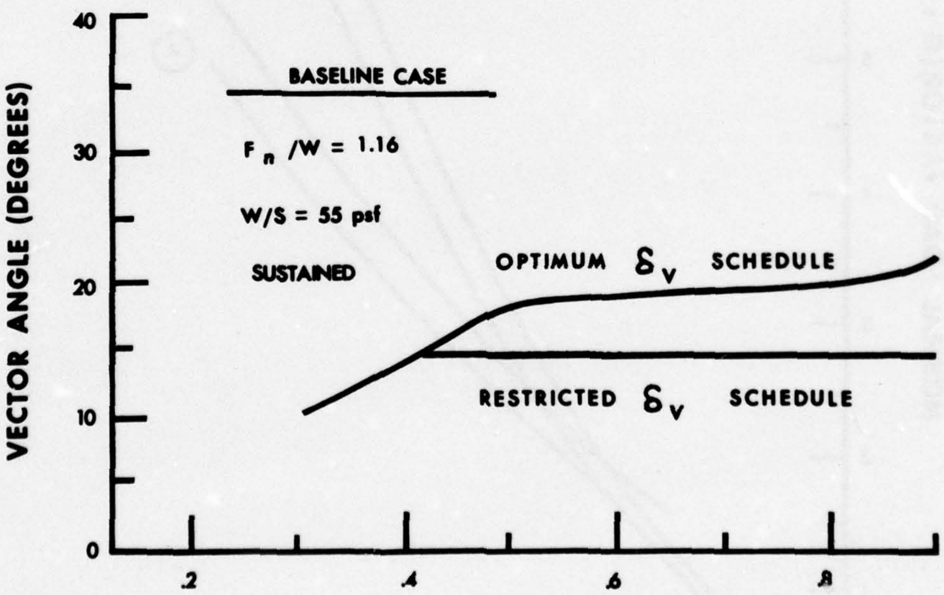


Fig.21b Thrust Vector Schedules - Sustained Turn

There are a number of points worth noting at this time. The optimum thrust vector angle is a function of where in the maneuvering envelope the aircraft is operating. Figures 20b and 21b depict the optimum thrust vector schedule for the previously discussed curves in the baseline case. For example at Mach .8, 20 degrees of vectoring are called for in the sustained case, and 34 degrees are required at the instantaneous limit. This represents a substantial vector capability. Since it is possible that nozzle design considerations might restrict the maximum vector angle, the increases in normal load factor achieved when the vector angle was limited to a maximum of 15 degrees were calculated and are shown in Figures 20a and 21a. Restricting the vector angle to a maximum of 15 degrees decreased the G payoff by about 36% for the instantaneous case and 12% for the sustained case. While considering payoffs due to thrust vectoring that do not result in a Ps penalty, it is worth noting that there are three different ways the results can be viewed: 1) at a given velocity, the addition of thrust vectoring can increase the maximum normal load factor capability at the same Ps; 2) a given load factor can be achieved at a lower velocity and the same Ps; 3) the same load factor can be achieved at the same velocity while experiencing an increase in Ps.

Sensitivity to Aircraft Design Variables

All of the above results were derived for an aircraft with a sea level static thrust loading of 1.16 and a combat wing loading of 55 lb/ft^2. As stated earlier, one of the objectives of the study was to examine whether the change in normal load factor, N, gained through the use of thrust vectoring is a function of basic design variables such as wing-loading and thrust loading. To determine the sensitivity of the payoff to these variables, four other cases beside the baseline aircraft were considered: aircraft with higher and lower thrust loadings (sea level static, Fn/TOGW = 1.28 and 0.68), and aircraft with lower and higher combat wing loadings (W/S = 50 and 90).

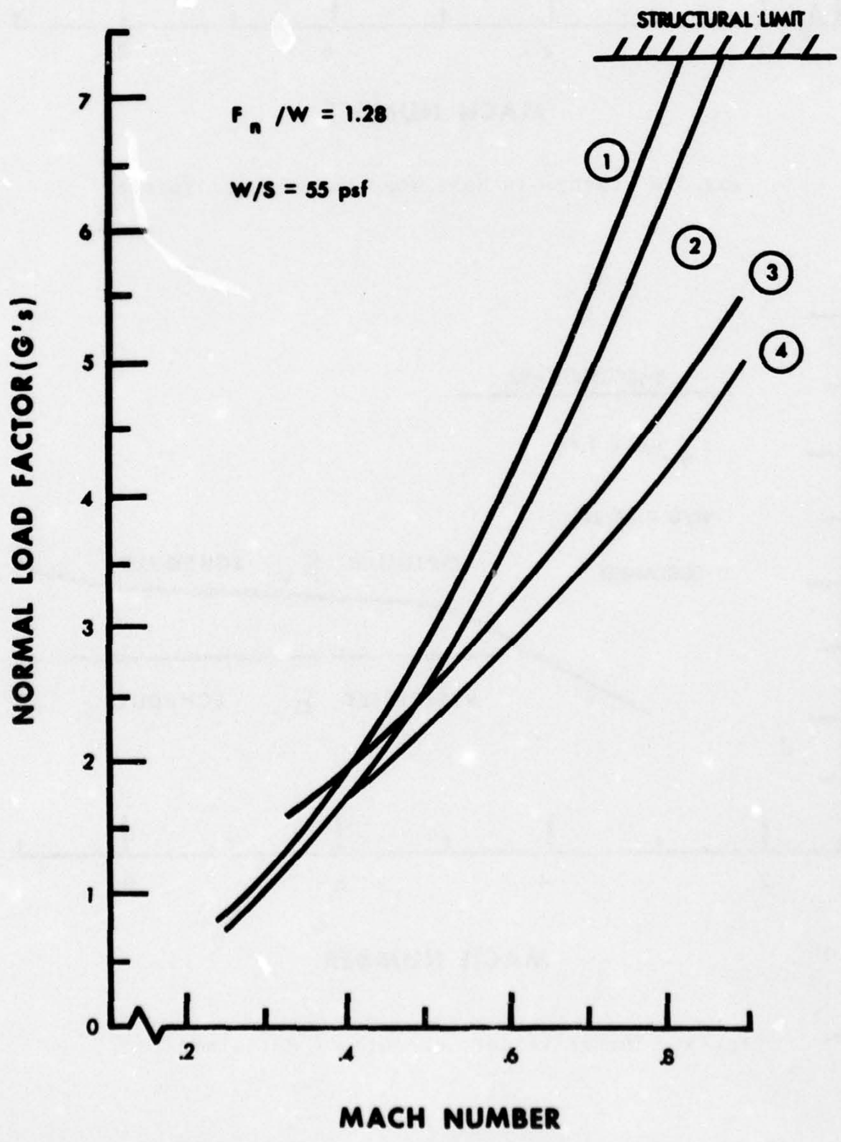


Fig.22 V-N Diagram - Increased Thrust Loading

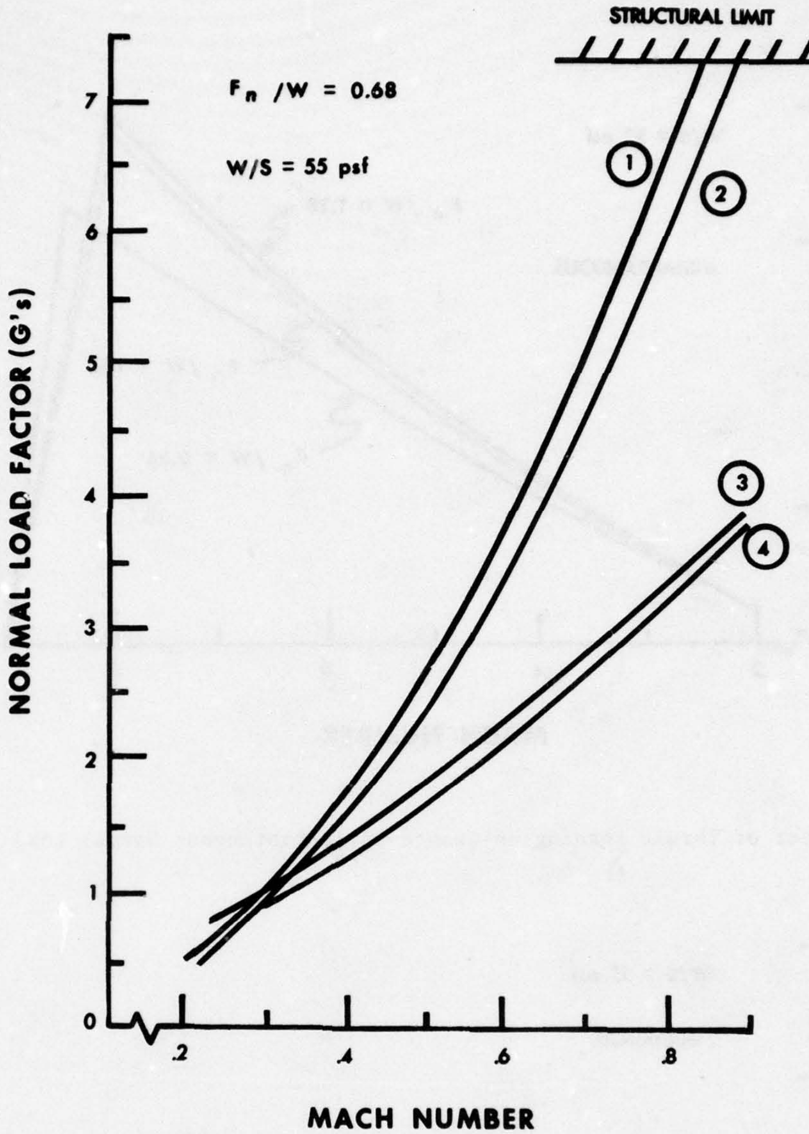


Fig.23 V-N Diagram - Decreased Thrust Loading

The results for a thrust loading of 1.28 are presented in Figure 22 and for a thrust loading of 0.68 in Figure 23. The four curves depicted in each figure were calculated the same way as described in the baseline case above. Consider the Δ normal load factor when thrust vectoring is added (i.e., the difference between curves 1 and 2 and between 3 and 4). To understand how this increased G capability varies with thrust loading, compare the normal load factor ,ΔN, due to thrust vectoring in these cases with the ΔN obtained in the baseline case where Fn/W = 1.16. Figures 24 and 25 will aid this comparison. In both figures, the ΔN due to thrust vectoring is depicted for thrust loadings of 0.68, 1.16 and 1.28. Note that an increase in thrust loading from 1.16 to 1.28 has very little effect on the instantaneous ΔN. Also note that the ΔN usually decreases to zero somewhere above Mach 0.8. This is due to the fact that the structural limit has been reached for the non-vectored case, and any additional increase in load factor is prohibited. If it were not for the structural limit, the ΔN due to vectoring would continue to increase. Since the structural limit has been reached, and additional lift is available due to vectoring, the aircraft angle-of-attack could be reduced, thus reducing induced drag and thereby increasing Ps. It is also evident from Figure 24 and 25 that the max sustained G capability is more sensitive to thrust loading than is instantaneous G capability. The sensitivity of ΔN payoff to thrust loading is more easily seen in the case of the more significant change in thrust loading achieved by reducing it to a value of 0.68. Note the magnitude of the sustained ΔN. At its maximum, the payoff is still less than .2G and this small payoff is further reduced as Mach number increases. The trend of decreased ΔN with reduced thrust loading makes sense. Since thrust is being vectored to produce a force in the lift direction, it is logical that less thrust available for vectoring would result in a smaller payoff.

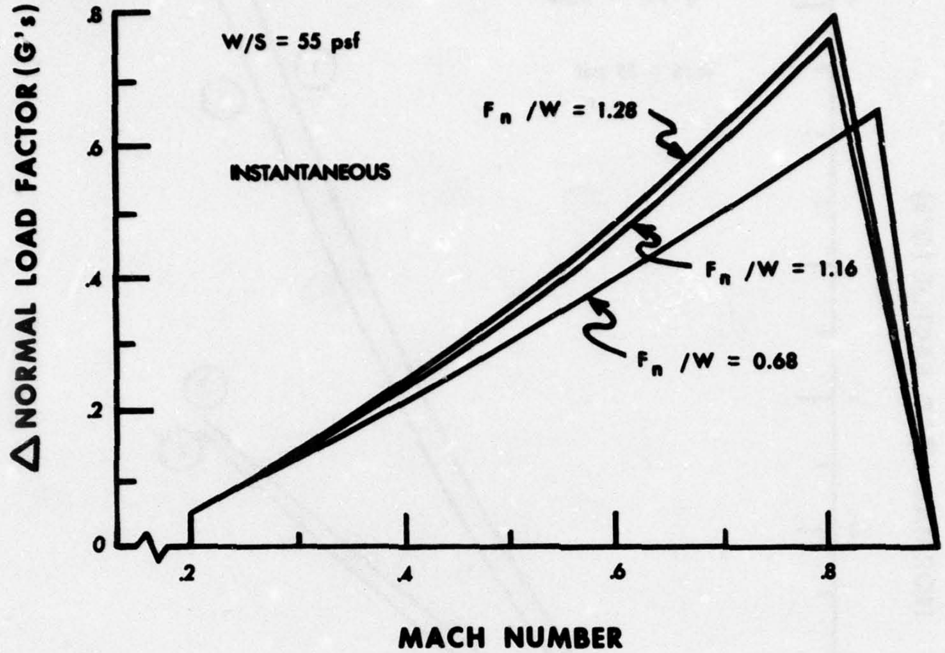


Fig.24 Effect of Thrust Loading on Change in Instantaneous Normal Load Factor

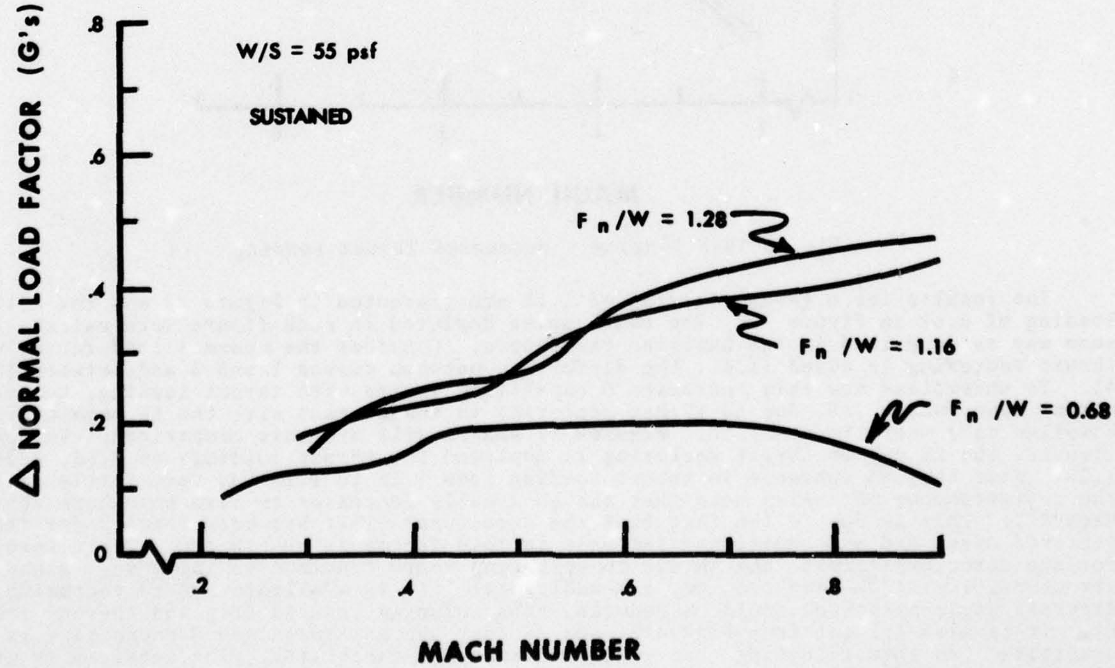


Fig.25 Effect of Thrust Loading on Change in Sustained Normal Load Factor

The baseline aircraft in this study had a combat wing loading of 55. In order to determine the sensitivity of thrust vectoring benefits to wing loading, cases were examined with wing loadings of 50 and 90. The results for wing loadings of 50 and 90 are presented in Figures 26 and 27 respectively and the resulting ΔN can be more closely examined in Figures 28 and 29. It is interesting to note that reducing wing loading from 55 to 50 results in almost the same instantaneous G capability without vectoring as the baseline

case with vectoring (compare Curve 1 on Figure 19 with Curve 2 on Figure 26). Comparing Curve 3 on Figure 19 with Curve 4 on Figure 26 shows that lowering wing loading to 50 without adding thrust vectoring capability achieves about half the normal load factor increase achieved by adding vectoring capability to the baseline aircraft. Figures 28 and 29 show that the ΔN to be gained is not significantly changed when reducing wing loading from 55 to 50. Note that this reduction in wing loading results in a very slight increase in instantaneous ΔN, but generally a slight decrease in sustained ΔN.

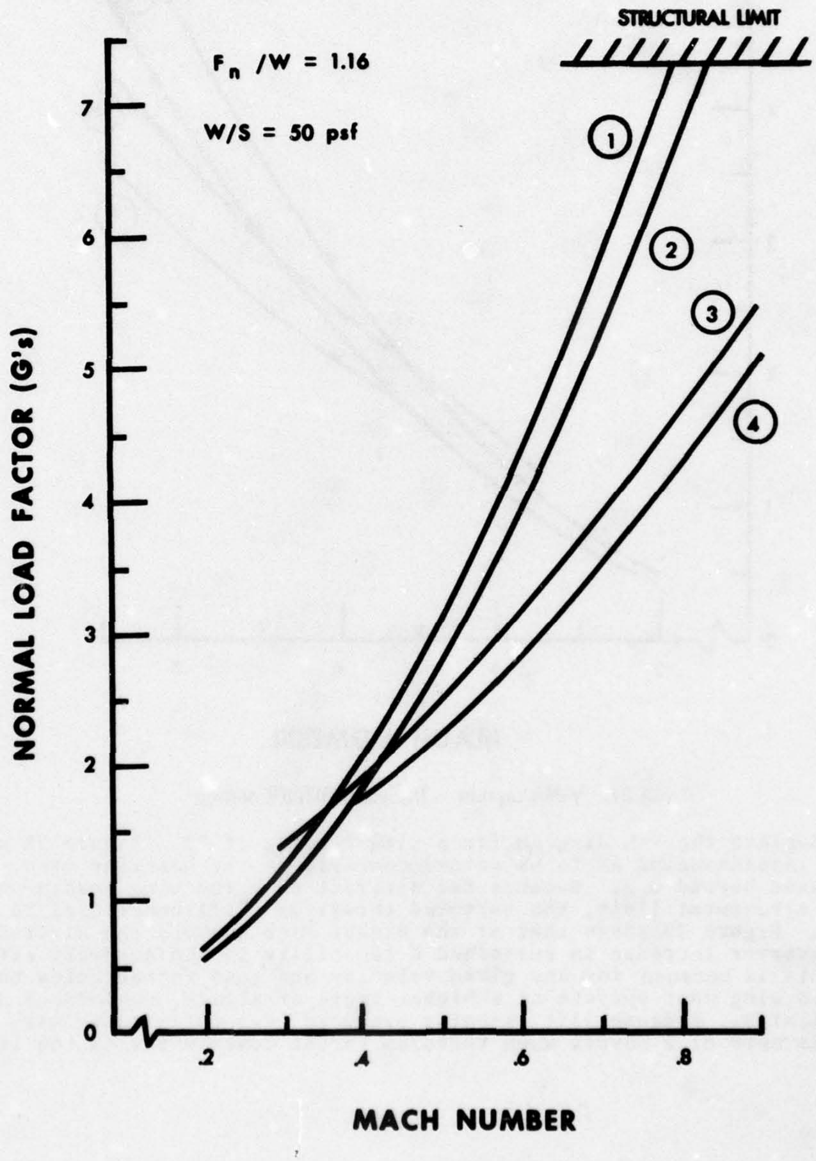


Fig.26 V-N Diagram - Decreased Wing Loading

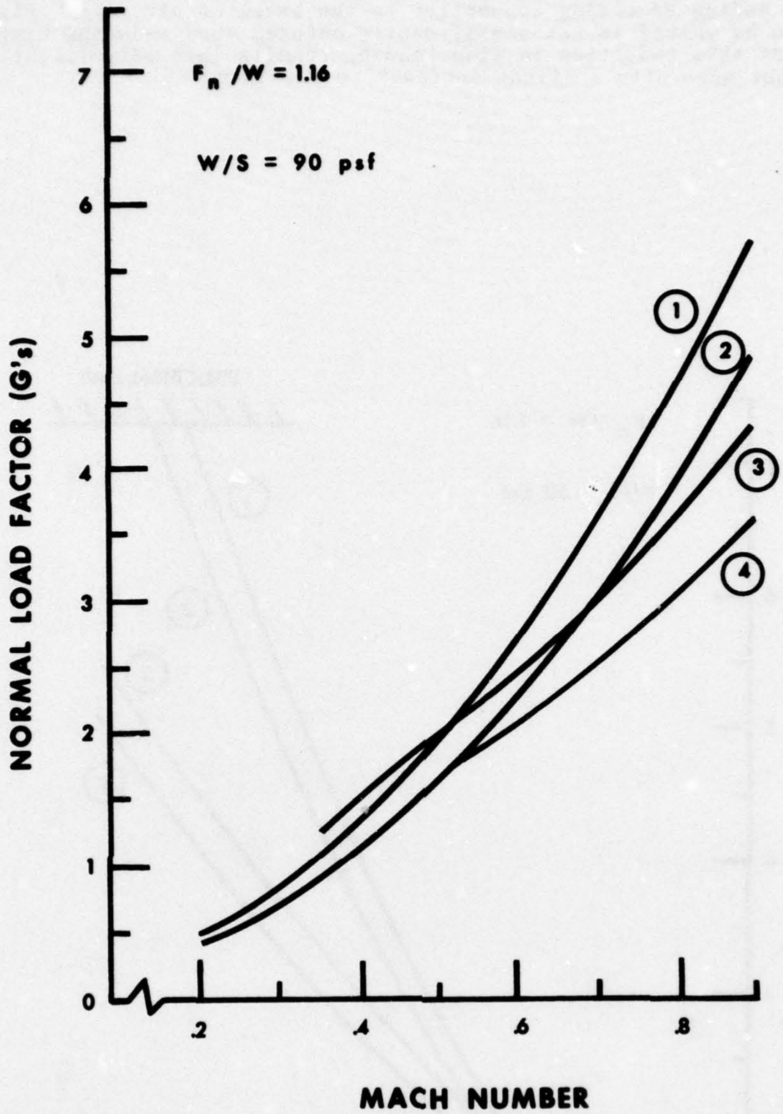


Fig.27 V-N Diagram – Increased Wing Loading

Figure 27 depicts the V-N diagram for a wing loading of 90. Figure 28 shows a slight decrease in the instantaneous ΔN to be gained compared to the baseline case, until Mach number is increased beyond 0.8. Because the aircraft with the wing loading of 90 has not yet reached its structural limit, the vectored thrust is still beneficial to increasing the load factor. Figure 29 shows that at the higher Mach numbers the aircraft that will experience the greater increase in sustained G capability is the aircraft with a highly loaded wing. This is because for any given velocity and load factor below the lift limit, the highly loaded wing must operate at a higher angle-of-attack, and induced drag is increased substantially. Because lift is being produced less efficiently with a high wing loading, there is more of a payoff when vectored thrust contributes to the lift.

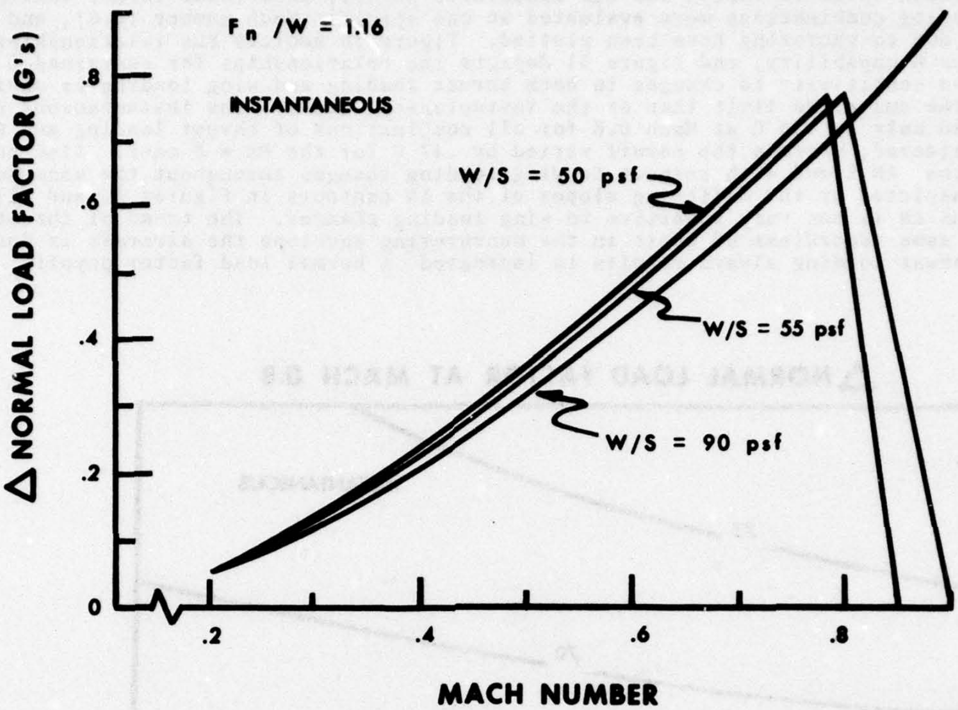


Fig.28 Effect of Wing Loading on Change in Instantaneous Normal Load Factor

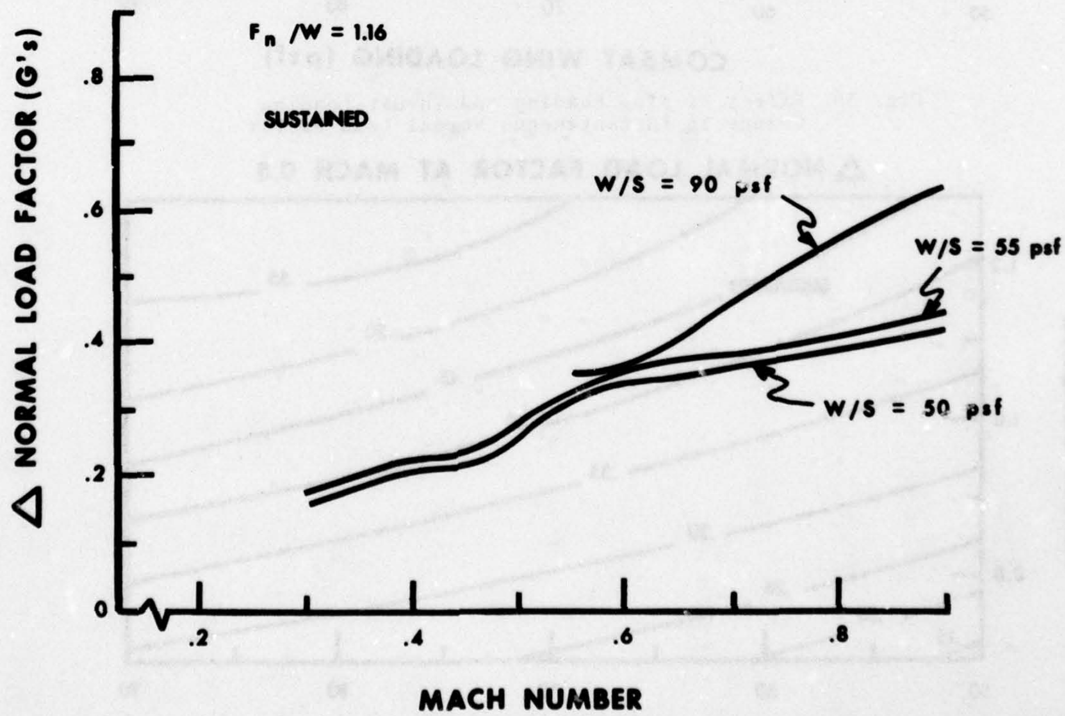


Fig.29 Effect of Wing Loading on Change in Sustained Normal Load Factor

The previous discussion has shown the relationship between the ΔN and changes in thrust loading and wing loading. One should recognize that the variations in thrust loading were evaluated at a relatively low wing loading, and the wing loading variations were evaluated at a relatively high thrust loading. In order to better define the relationship between these variables and the associated payoff, additional thrust loading and wing loading combinations were evaluated at one specific Mach number (0.8), and contours of ΔN due to vectoring have been plotted. Figure 30 depicts the relationships for instantaneous G capability, and Figure 31 depicts the relationships for sustained G capability. The sensitivity to changes in both thrust loading and wing loading is obviously greater at the sustained limit than at the instantaneous limit. The instantaneous ΔN payoff varied only by .20 G at Mach 0.8 for all combinations of thrust loading and wing loading considered, whereas the payoff varied by .47 G for the Ps = 0 case. Also note that while the ΔN trend with respect to wing loading changes throughout the maneuver envelope (as depicted by the differing slopes of the ΔN contours in Figures 30 and 31), the instantaneous ΔN is not very sensitive to wing loading changes. The trend of thrust loading remains the same regardless of where in the maneuvering envelope the aircraft is operating. Increased thrust loading always results in increased Δ normal load factor payoff.

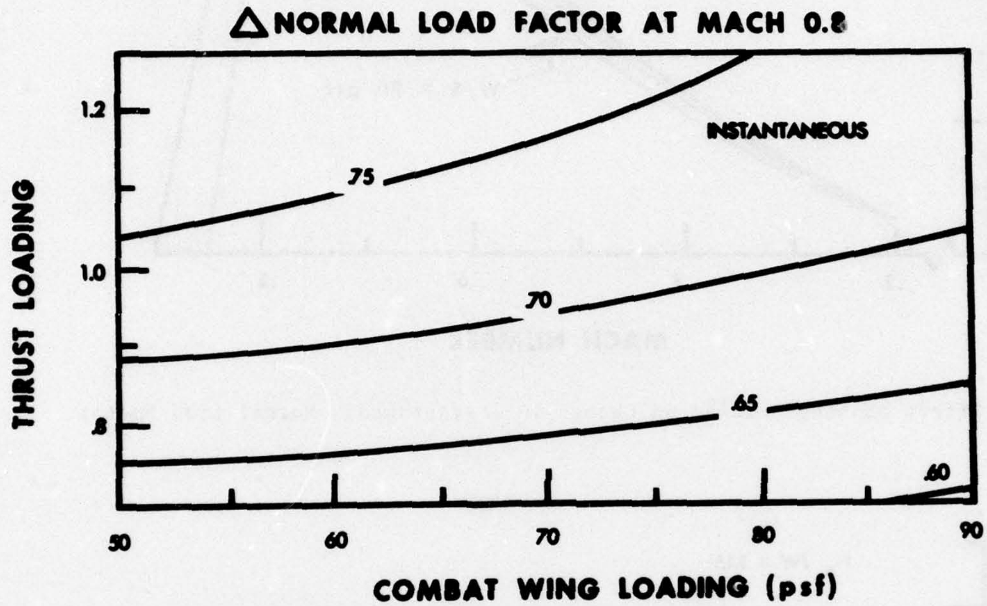


Fig. 30 Effect of Wing Loading and Thrust Loading Change in Instantaneous Normal Load Factor

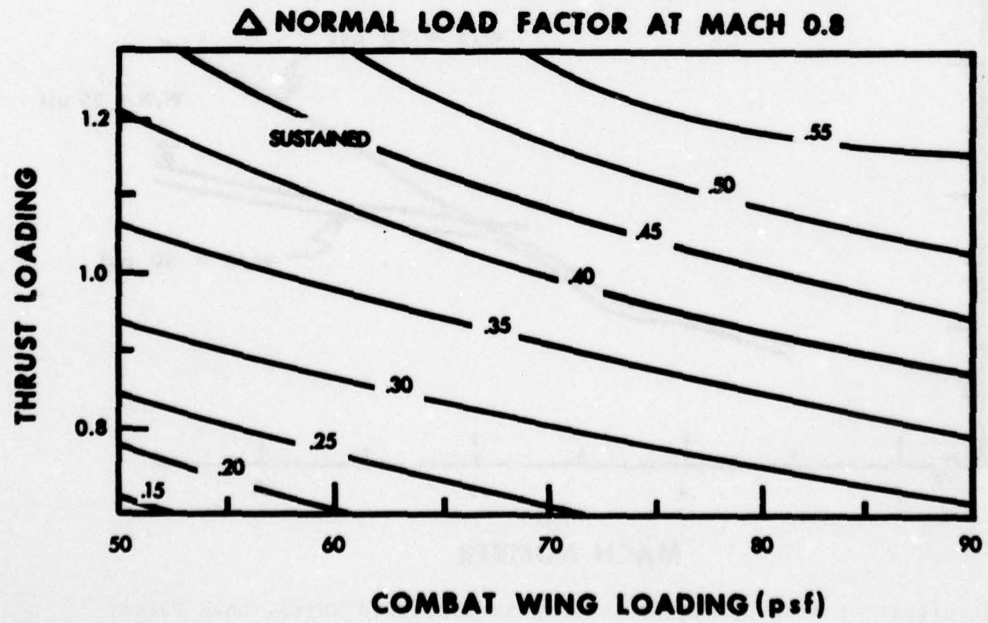


Fig. 31 Effect of Wing Loading and Thrust Loading - Change in Sustained Normal Load Factor

Figures 32 and 33 depict the optimum thrust vector angles corresponding to the ΔN payoffs at Mach .8 depicted in Figures 30 and 31, respectively. Note that for maximum instantaneous G capability, very large vector angles are required (up to 50 degrees). In order to determine how dependent this payoff is on achieving these large vector angles, the ΔN payoff for 15 degrees of vectoring was examined. Limiting vector angle to 15 degrees would reduce the ΔN payoffs in Figure 20a by a maximum of .33 G's and a minimum of .21 G's. This would represent a relatively significant decrease in payoff. Figures 33 and 21b show that smaller vector angles are required at the maximum sustained limit (optimum deflection angle ranged from 15 to 28 degrees). Since the optimum deflection angle is smaller for the maximum sustained case, restricting the deflection to 15 degrees had a much smaller effect. The ΔN payoffs depicted in Figure 21a would be decreased by a maximum of .13 G's and a minimum of 0 G's.

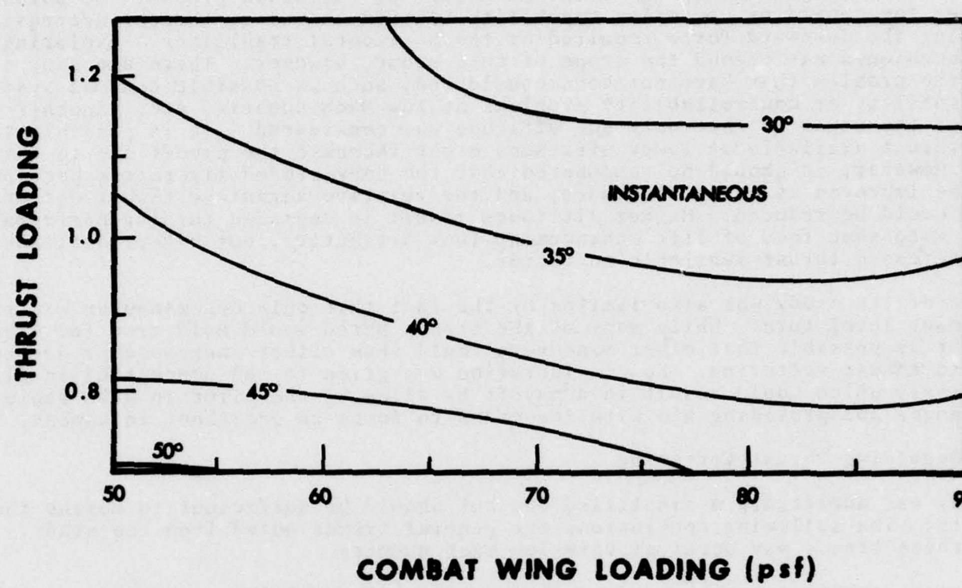


Fig.32 Effect of Wing Loading and Thrust Loading on Vector Angle in Instantaneous Turn

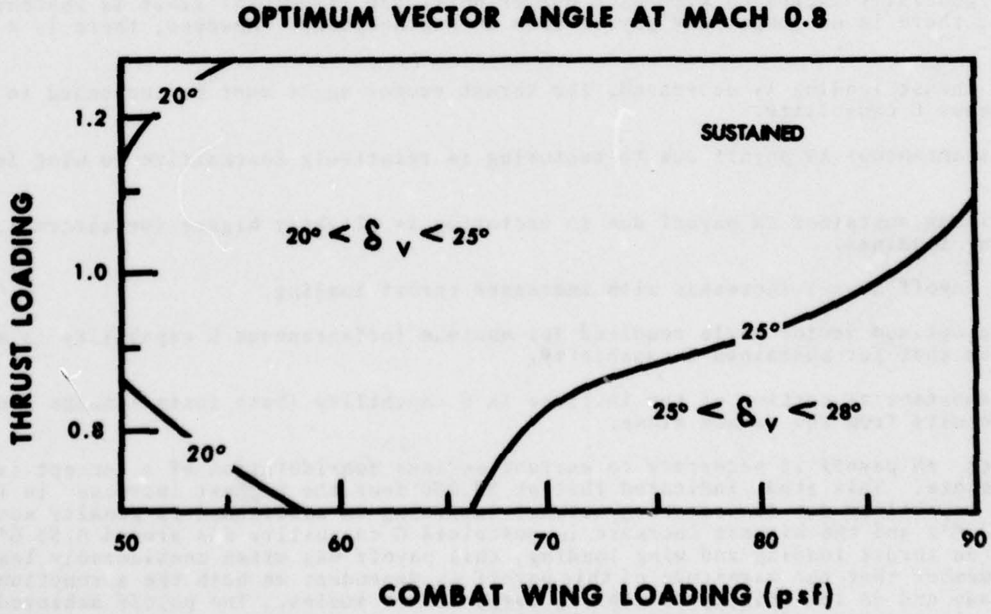


Fig.33 Effect of Wing Loading and Thrust Loading on Vector Angle in Sustained Turn

Limitations of the Analysis

Some of the assumptions made in this study deserve discussion at this time. The assumption of no weight penalty due to adding thrust vectoring capability and a canard obviously is optimistic. There naturally will be a weight penalty involved. The result will be a slightly smaller ΔN payoff due to thrust loading and wing loading changes. Range would also be slightly decreased. The assumption of no canard-wing interaction is also optimistic, and any aerodynamic interference between the canard and the wing would reduce the payoff. The assumption of no nozzle induced drag reduction would probably have a very small effect. This is because any resulting drag reduction would most likely represent a small fraction of the total drag at the high angles of attack encountered in the study. The effect of assuming no nozzle induced lift enhancement is unknown. One of the few pessimistic assumptions was that of trim technique. It was assumed that the wing induced pitching moment would be trimmed with the horizontal tail and that the pitching moment induced by thrust vectoring would be trimmed with the canard. It is recognized that this is not the optimum way to trim the aircraft. It would probably be advantageous to use either the canard or vectoring capability to trim the wing, thereby decreasing or eliminating the downward force required by the horizontal stabilizer. Exploring alternate trim techniques was beyond the scope of this study, however. There are many other aspects of the problem that have not been considered, such as possible control system problems, stability or controllability problems at low Mach numbers, etc. Another limitation of the study is that only one altitude was considered. It is possible that the higher thrust available at lower altitudes might increase the payoff due to thrust vectoring. However, it should be remembered that the nonvectored aircraft's performance would also be improved at lower altitudes, and the relative advantage thrust vectoring might offer could be reduced. Higher altitudes result in degraded turning performance, which might make some form of lift enhancement look attractive, but higher altitudes also result in decreased thrust available to vector.

The scope of the study was also limited by the fact that only one maneuver was studied - a maximum power level turn. While many of the trends noted would hold true for other maneuvers, it is possible that other maneuvers could show either increased or decreased payoff due to thrust vectoring. No consideration was given to the concept of in-flight thrust reversal, which could result in a payoff by allowing the pilot to make rapid airspeed changes and providing him with the means to force an overshoot in combat.

Conclusions Regarding Thrust Vectoring

This study was admittedly a simplified one but should be sufficient to define the first order effects. The following conclusions are general trends noted from the study. Deviations from these trends may occur at very low Mach numbers.

1. Thrust vectoring enhances maximum instantaneous G capability more than maximum sustained G capability. However, the substantially greater benefit at the lift limit is highly dependent on the ability to achieve large vector angles.

2. Sustained G capability is more sensitive than instantaneous G capability to changes in either thrust loading or wing loading.

3. ΔN generally increases with Mach number until the structural limit is reached. At this point, there is no longer any payoff from a ΔN standpoint. However, there is a small Ps payoff.

4. As thrust loading is decreased, the thrust vector angle must be increased to maximize instantaneous G capability.

5. Instantaneous ΔN payoff due to vectoring is relatively insensitive to wing loading changes.

6. Maximum sustained ΔN payoff due to vectoring is slightly higher for aircraft with higher wing loadings.

7. ΔN payoff always increases with increased thrust loading.

8. The optimum vector angle required for maximum instantaneous G capability is much higher than that for sustained G capability.

9. A substantial portion of the increase in G capability (both instantaneous and sustained) results from the canard alone.

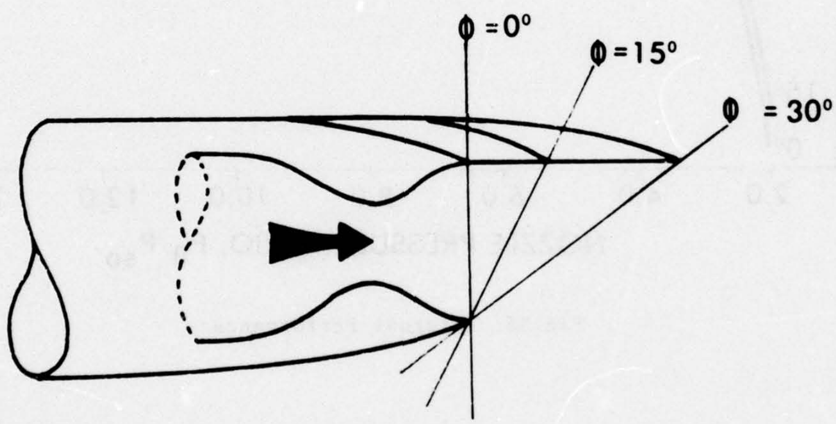
How much ΔN payoff is necessary to warrant serious consideration of a concept is subject to debate. This study indicated that at 30,000 feet the highest increase in instantaneous G capability due to vectoring without incurring an additional Ps penalty was around 0.8 G's and the highest increase in sustained G capability was around 0.55 G's. Depending on thrust loading and wing loading, this payoff was often considerably less. One should remember that the magnitude of this payoff is dependent on both the assumptions previously made and on the ability to achieve large vector angles. The payoff achieved in actual practice could quite possibly be smaller.

Before drawing any conclusions on the relative value of thrust vectoring, other alternatives should be examined. In this study, the lift from the canard was responsible for about 2/3 of the increase in G capability. This fraction would naturally change with changes in canard design and location. But practical considerations place restrictions on canard design and placement, and it is easy to conclude that the canard lift is likely to be at least of the same magnitude as the force resulting from thrust vectoring. For this reason, it would be wise to explore the option of simply adding a canard without any thrust vectoring capability. The canard could be used to trim the wing pitching moment and thus reduce the downward force produced by the horizontal stabilizer. With a canard and horizontal stabilizer both operating at the proper positive angle of attack, the aircraft could be trimmed and at the same time experience a normal load factor payoff resulting from the lift produced by the canard and horizontal stabilizer. The net result would be to effectively decrease the wing loading of the aircraft (by supplementing lift with canard and horizontal stabilizer lift). This suggests another alternative which should be considered: decreasing wing loading by increasing the size of the wing. Obviously the selection of the desired wing loading is affected by many factors in addition to desired turning performance. But it is important to weigh any benefits and penalties associated with concepts which effectively reduce wing loading against the benefits and penalties associated with simply increasing wing size. The additional complexity associated with sophisticated concepts of lift enhancement may result in payoffs which are comparable or only marginally better than that which could be obtained using more conventional methods.

In order to properly compare alternate concepts with thrust vectoring, additional study is needed. Optimum trim methods need to be determined, the effects of canard-wing interaction need to be studied, etc. It is possible that thrust vectoring might be desirable for reasons other than improved maneuverability, for example, reduction of infrared signature. All of these factors need to be properly evaluated.

The most important point to remember is that whenever a concept is selected, by necessity, other concepts and options are excluded. A concept should be selected only after determining that it will result in the greatest overall payoff for the time, effort and money to be expended.

FORCE ACCOUNTING PROCEDURES



FORCE APPLIED NOT NECESSARILY EQUAL TO $\dot{M}_9 V_9 + (Ps_9 - Pso)\, A_9$

Fig.34 Family of Single Ramp Nozzles

Traditionally, it has been possible to treat gross thrust as a vector whose orientation was aligned with the axis of the nozzle and remained so aligned during variations in nozzle pressure ratio. For some nonaxisymmetric nozzle types and vectoring schemes, such a treatment of gross thrust is not valid. As an example, consider a family of single ramp expansion nozzles formed by extending an axisymmetric or two-dimensional configuration as depicted in Figure 34. While the momentum terms of all of those nozzles are aligned with the nozzle axis, the direction of the pressure area term varies with the orientation of the exit plane, and therefore the magnitude and direction of the gross thrust vector varies with the nozzle pressure ratio. It is only because the vector representing the pressure area term and the vector representing the momentum term have traditionally been co-linear that in the past it has been possible to reduce the vector expression for gross thrust to a single vector invariant in direction. Figure 35 and 36 depict how gross thrust varies in magnitude and direction as a function of nozzle pressure ratio, respectively, with changes in exit plane orientation. These calculations were based on the ideal case of constant internal area ratio, no internal losses, and an internal contour which results in horizontal flow at the exit. Note that the force applied in the axial or horizontal direction is independent of exit

area orientation, and at the design pressure ratio the total force applied is independent of exit area orientation. However, at all other pressure ratios, the magnitude and direction of the applied force is sensitive to exit area orientation. The trends shown here are not limited to the case of the single ramp nozzle. Similar problems will be encountered with any nozzle design that does not maintain the exit area normal to the flow direction. This is true for nearly all vectoring schemes and particularly true for plug type nozzles or designs using louvers at the nozzle exit to turn the flow. This potential problem does not necessarily have a negative effect, however. If properly integrated and accounted for, changes in force magnitude and direction can result in benefits to the system. The basic problem arises in situations where a proper accounting of the forces is not made resulting in erroneous conclusions being drawn. Caution is particularly important in the formulation of wind tunnel tests to determine the external installation benefits attributable to nonaxisymmetric nozzle types. In such tests and in system studies it is best to treat the gross thrust as a vector quantity whose orientation may be a function of nozzle pressure ratio as well as nozzle deflection angle.

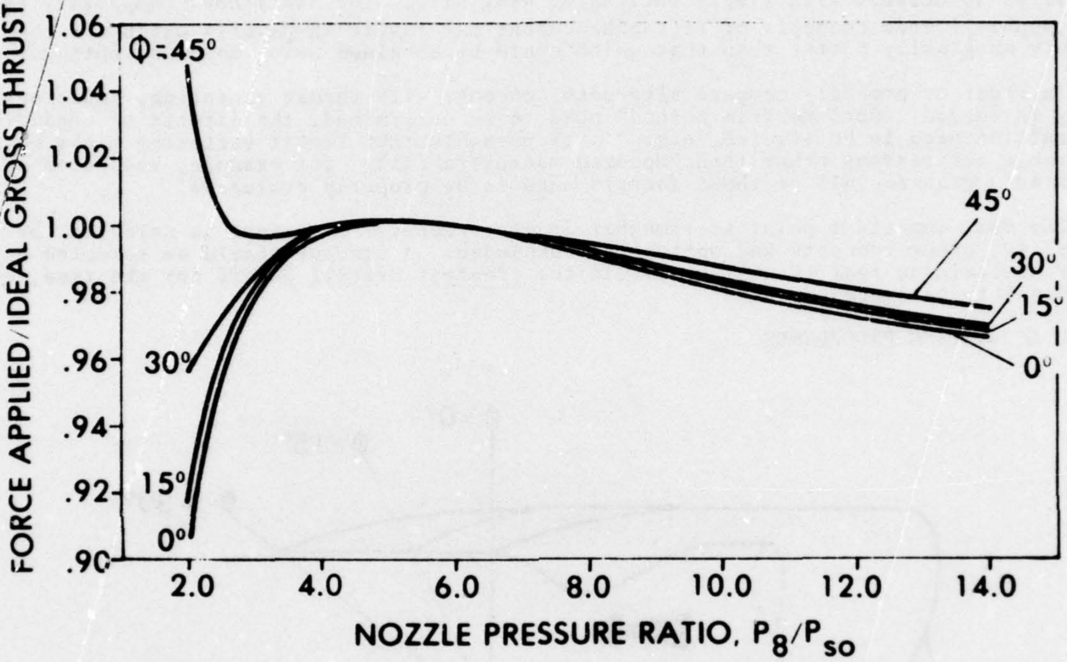


Fig.35 Internal Performance

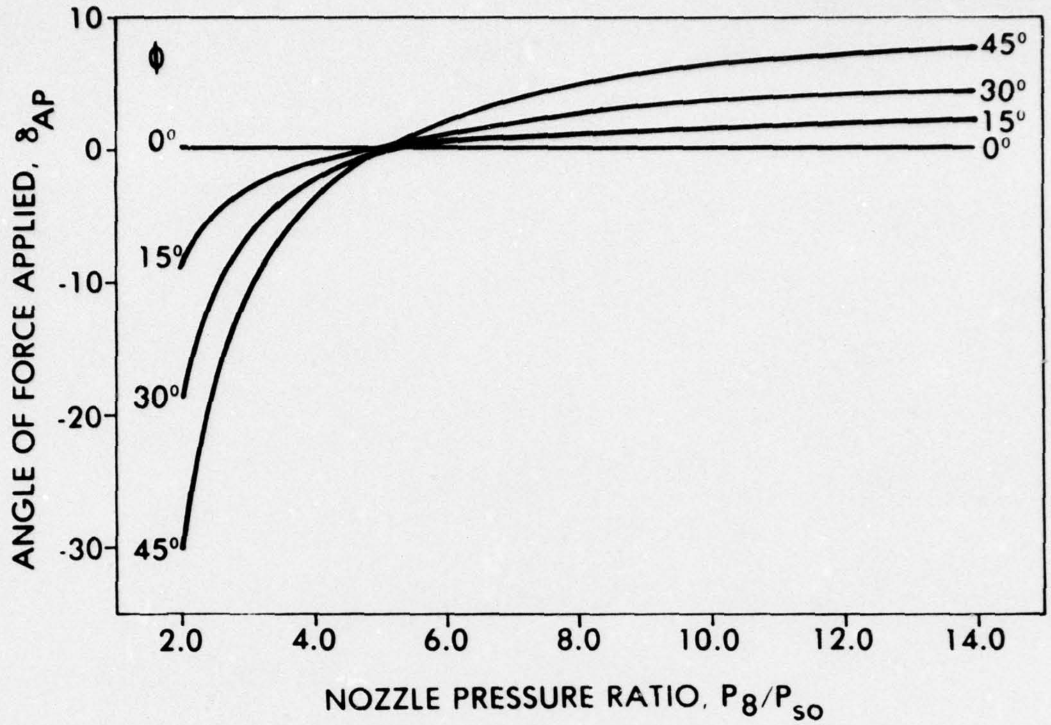


Fig.36 Angle At Which Thrust Is Applied

CONCLUSIONS

Advanced nozzles offer a number of potential benefits to future aircraft. These benefits have been discussed and related to specified levels of internal nozzle performance, cooling effectiveness, and structural or weight considerations. The installation benefits which must be achieved to offset the increased weight of these nozzles to make them competitive with more conventional configurations have been discussed. The value of thrust vectoring has been considered and related to basic aircraft parameters. Lastly, some potential problem areas in the force accounting system for nonaxisymmetric nozzles have been highlighted.

There are a number of areas which require further development effort. It is important to agree upon a system of force accounting for these types of nozzles. An installation data base must be developed, and it may be necessary to refine or develop new test techniques to provide the necessary wind tunnel data. A full-scale ground-based demonstration of the advanced cooling techniques and the structural integrity of these advanced concepts is needed. Development of the stability and control aspects of engines using nonaxisymmetric nozzles is essential. Particular attention must be given to thrust vectoring effects since its greatest potential seems to lie in an area where distortion tolerance, engine stability and augmentor ignition are most difficult to achieve. Control technology needs to be developed so as to properly integrate the engine and nozzle control and define the appropriate interfaces with the aircraft flight control systems. Further development in all of these areas is required before these advanced nozzle concepts can be considered as design options of acceptable risk.

ACKNOWLEDGEMENTS

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LA RECHERCHE D'UN COMPROMIS ENTRE LA COMPLEXITE D'UN MOTEUR DE CHASSEUR ET SON COUT

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INTRODUCTION.

- Le problème du choix de la complexité d'un moteur se pose au motoriste à trois occasions :
  . pour prévoir l'évolution des moteurs,
  . quand il veut lancer un nouveau moteur,
  . quand, au cours de la phase projet d'un avion, il veut influer sur le choix du moteur.

- Le degré de complexité retenu comptera directement pour :
  . l'ampleur des frais de développement,
  . la facilité d'emploi par l'utilisateur,
  et, d'une façon moins évidente sur :
  . le coût de l'avion.

Le motoriste doit s'assurer que dans le choix du degré de complexité, on n'a pas pénalisé trop fortement les deux premiers points, en surestimant le gain sur le troisième point.

Il doit donc pouvoir estimer le coût de l'avion.

1. BUT DE L'ETUDE.

Le coût de l'avion est souvent relié à la masse au décollage. Or, la complexité du moteur intervient sur le coût de l'avion :
. par les performances du moteur qui peuvent, à mission donnée, réduire la masse de l'avion,
. mais aussi par le coût du moteur lui-même.

Il était également intéressant de faire intervenir simultanément ces deux effets.

Dans la présente étude, j'examine donc le résultat d'une minimisation du coût, que je confronte à celui d'une minimisation de la masse, en fonction des caractéristiques du moteur, et ceci pour plusieurs missions dans lesquelles des performances différentes du moteur sont mises en valeur.

2. METHODE UTILISEE POUR L'ETUDE.

J'ai utilisé, pour cette étude, un modèle d'avion destiné à calculer la masse d'un avion remplissant une mission donnée avec un moteur donné.

J'ai défini une famille de moteurs de cycles différents, mais de même niveau technique.

J'ai calculé, pour chaque moteur de la famille, la masse au décollage minimale de l'avion satisfaisant la mission, puis son coût.

2.1. Modèle de l'avion - Planche 1.

Ce modèle est simplifié, mais permet :
- de définir pour chaque mission un avion correctement adapté au domaine de vol parcouru,
- de tenir compte de l'installation du moteur et de la capacité de carburant.

Pour l'adaptation au domaine de vol, il comprend les paramètres suivants :
- la charge alaire m_o/s
- la flèche φ
- l'épaisseur relative e
- le type d'entrée d'air, subsonique ou supersonique,
- l'adaptation de l'entrée d'air à un point donné du domaine de vol.

Pour l'adaptation au moteur, il comprend les paramètres suivants :
- *dimensions caractéristiques* du fuselage dépendant du diamètre du moteur, $\emptyset$
- masse du moteur m

Enfin, une dimension caractéristique supplémentaire du fuselage est prise en compte pour tenir compte de la capacité carburant, L.

Il fournit, en fonction de ces paramètres :
- un devis de masse, en particulier la masse à vide : m vide
- la masse de carburant logeable : m^{*}_{CO}
- la traînée en fonction de la portance le long du profil du vol
- les corrections à apporter à la poussée du moteur :
  . rendement d'entrée d'air
  . traînée additive.

2.2. Modèle du moteur - Planche 2.

Le modèle retenu pour la famille de moteurs correspond à un double flux à flux mélangés, fan pur, ce qui permet de se limiter à quatre paramètres :
- rapport de pression π
- taux de dilution λ
- température devant turbine T_5
- post-combustion ou non.

On a retenu :
- 4 valeurs de π : 10, 15, 20, 30
- 4 valeurs de λ : 0,2 0,5 1 2
- 2 valeurs de T_5 : 1300°C, 1400°C
- la post-combustion ou non

ce qui a conduit à faire l'avant-projet détaillé de 32 moteurs, chacun avec ou sans post-combustion.

Le niveau technique est le même pour tous les moteurs. Une limitation de pression interne maximale à grande Vi intervient pour les moteurs les plus comprimés.

On a calculé :
- les performances dans tout le domaine de vol,
- la masse et l'encombrement,
- le coût.

Ce dernier est évalué à partir de la poussée et d'un degré de complexité qui dépend principalement du nombre et de la complexité des étages de compression et de détente.

Un exemple de performance (consommation spécifique) qui varie différemment avec π et λ, suivant le cas de vol, est donné planche 3 :
- consommation spécifique en sec à M = 0,9 Z = 0
- consommation spécifique en PC à M = 0,8 Z = 20.000 ft.

2.3. Calcul de la masse et du coût - Planche 4.

- La mission est définie comme une suite de segments.
- On calcule la masse minimale, pour chaque cycle moteur.
- Dans le modèle de l'avion, les paramètres d'entrée "MISSION" sont fixés à des valeurs plausibles pour la mission considérée. Seuls restent variables :
  . les paramètres "MOTEUR" $\emptyset$ et m, qui, pour un cycle donné, sont tous deux reliés à la poussée maximale au point fixe F_o,
  . les paramètres "TAILLE" m_o (masse au décollage) et L_4 (fixant la capacité de carburant).
- Dans le modèle du moteur, seule reste variable la taille du moteur F_o.
- Pour un jeu des trois paramètres variables, F_o, m_o, L_4, les modèles moteur et avion donnent :
  . une sortie "performances"
  . une sortie "devis de masses".

A partir de la sortie "PERFORMANCES", le calcul de la mission est effectué segment par segment, et on aboutit à trois sorties :
  . la poussée nécessaire à chaque segment F_j
  . la masse finale m' vide
  . la masse de carburant m_{CO}

On itère sur les trois paramètres d'entrée jusqu'à ce que :
  . la poussée F_j nécessaire à au moins un segment de la mission soit la poussée maxi dans ce cas de vol F^{*}_j
  . la masse finale m' vide soit la masse à vide du devis de masse m vide
  . la masse de carburant m_{CO} soit inférieure ou égale à la masse de carburant logeable du modèle m^{*}_{CO}.

- Les sorties "MASSES" sont alors introduites dans un modèle des coûts simplifié :
  . coût de la cellule = $f(m_o)$
  . coût du moteur = $f(F_o$, indice de complexité)
  . coût des équipements = Cte

On admet, et c'est pratiquement le cas, que le coût de l'avion de masse minimale est le coût minimal.

- Le même calcul est répété avec les 32 moteurs, ou les 64 (avec et sans PC) si le choix entre moteur sec ou moteur PC n'est pas évident.

3. ANALYSE DE QUATRE MISSIONS.

Le modèle permet d'étudier des missions de tout type.

J'ai étudié quatre missions :
- pénétration basse altitude,
- appui tactique,
- supériorité aérienne,
- interception,

en choisissant les durée des segments caractéristiques de façon à mettre en valeur telle ou telle qualité du moteur.

La discussion ne portera que sur les variations de π et de λ, qui sont les deux paramètres les plus significatifs de la complexité, à niveau technologique donné. Le fait de n'avoir que deux paramètres permet une représentation plus aisée.

3.1. Pénétration basse altitude.

1) Caractéristiques succinctes de la mission :

- Armement : 1500 kg
- Rayon d'action : 800 km, dont 1/3 à M = 0,9
- Combat : 5 mn à 3 g
- Décollage : en 1500 m

2) Caractéristiques de l'avion :

- $\frac{m_o}{s}$ = 700 kg/m^2 $\varphi = 45°$ e = 8 %

3) Résultats - Planche 5

Les avions sont légers et peu motorisés.
Les moteurs doivent avoir la PC, qui est utilisée au décollage et en combat.

On a porté en ordonnée sur la planche 5, rapportés aux valeurs minimales, la masse et le coût de l'avion en fonction de λ et π.

En prenant pour critère de ne pas dépasser de 2 % le minimum, on voit que l'optimisation en coût laisse :
- une grande liberté de taux de compression : $10 < \pi < 30$
- une faible liberté de taux de dilution : $0,2 < \lambda < 0,8$

alors que l'optimisation en masse pourrait laisser croire à une très faible liberté en taux de compression : $22 < \pi < 30$.

4) Conclusion.

Un moteur peu dilué et moyennement comprimé serait aussi bien adapté à cette mission qu'un moteur très comprimé, tout en étant moins complexe.

Les considérations de coût de développement et de facilité d'emploi pourraient faire pencher en sa faveur.

3.2. Appui tactique.

1) Caractéristiques succinctes de la mission :

- Armement : 4000 kg
- Rayon d'action : 400 km en croisière économique
- Appui : 10 mn à n = 4 avec demi charge utile
- Décollage : 1200 m

2) Caractéristiques de l'avion :

- $\frac{m_o}{s}$ = 500 kg/m^2 $\varphi = 35°$ e = 10 %

3) Résultats - Planche 6

Les avions sont lourds et peu motorisés.
Les moteurs doivent être sans PC (ce qui s'explique par la très longue durée du segment Appui).
Là encore, l'optimisation en coût
- impose un faible taux de dilution : $0,2 < \lambda < 0,6$
- laisse une grande liberté de taux de compression : $10 < \pi < 30$,

alors que l'optimisation en masse restreignait la gamme de taux de compression : $16 < \pi < 30$.

4) Conclusion.

Là encore, un moteur moyennement comprimé pourrait être préféré à un moteur très comprimé.

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FIGHTER AIRCRAFT DESIGN.(U)

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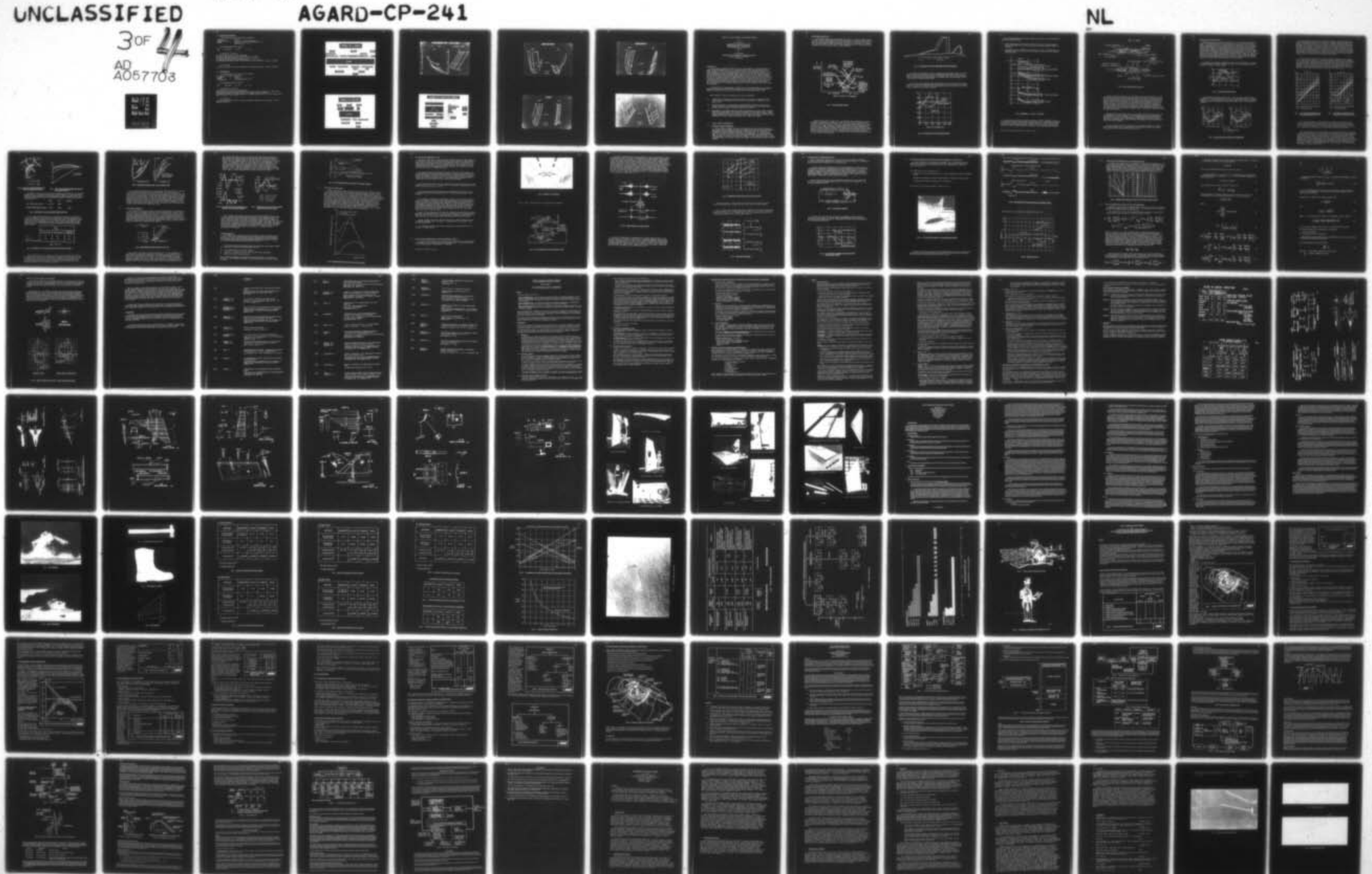
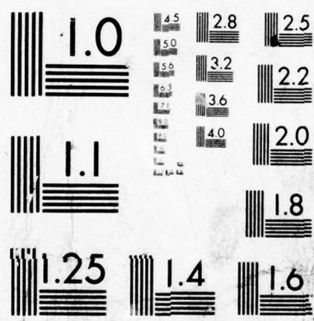



1.0
1.1
1.25
1.4
1.6
1.8
2.0
2.2
2.5
2.8
3.2
3.6
4.0

3.3. Supériorité aérienne.

1) Caractéristiques succinctes de la mission :

- Armement	:	300 kg
- Rayon d'action	:	400 km en croisière économique
- Combat	:	5 mn à n = 4 g à 20.000 ft, M = 0,8
		Capacité de n = 5 g

2) Caractéristiques de l'avion :

- $\frac{m_o}{s}$ = 300 kg/m² $\varphi = 50°$ e = 4 %

3) Résultats - Planche 7

Les avions sont légers et très motorisés.
Le moteur doit avoir la PC, et l'utiliser pour le combat.
Le taux de dilution doit être faible : $0,2 < \lambda < 0,5$.
Le taux de compression peut être quelconque : $10 < \pi < 30$.
L'optimisation en masse aurait été beaucoup plus contraignante : $0,2 < \lambda < 0,25$
$18 < \pi < 30$

4) Conclusion.

Là encore, le moteur à taux de compression très élevé n'est pas la seule solution.

3.4. Interception.

1) Caractéristiques succinctes de la mission :

J'ai choisi une mission très dure :

- Armement	:	300 kg
- Montée	:	60.000 ft, M = 2 en 3 '
- Poursuite de	:	2' à 60.000 ft, M = 2

2) Caractéristiques de l'avion :

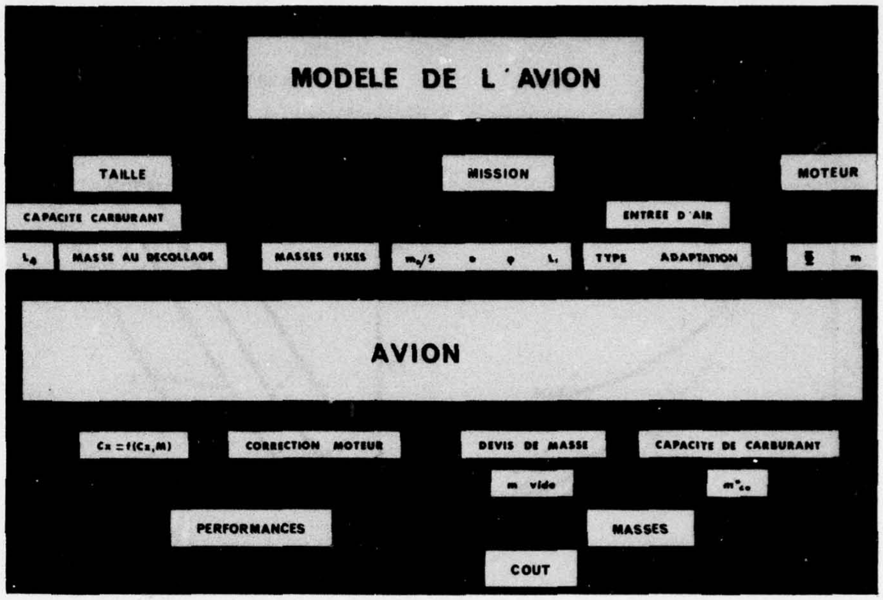
- $\frac{m_o}{s}$ = 300 kg/m² $\varphi = 50°$ e = 4 %

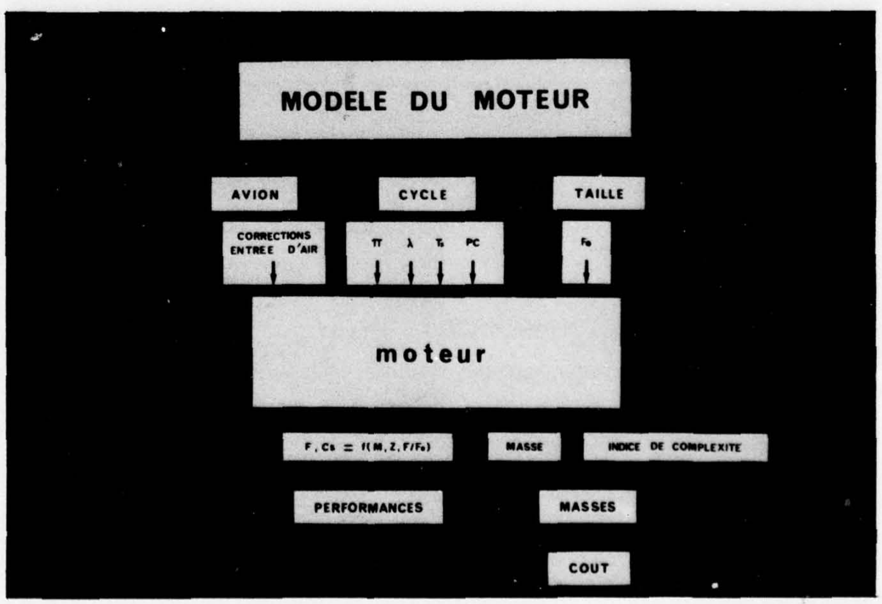
3) Résultats - Planche 8

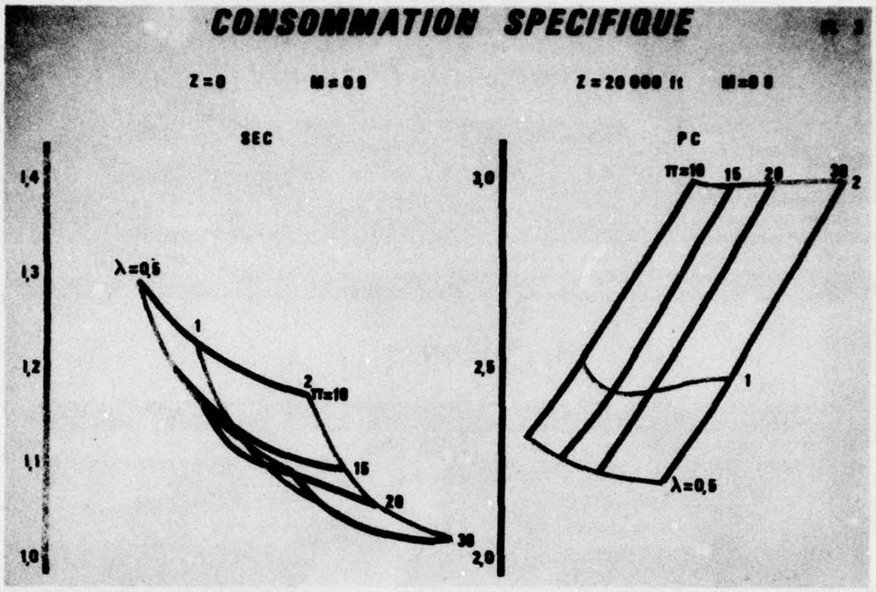
Les avions sont légers et très motorisés.
L'optimisation en coût donne les mêmes résultats que celle en masse, car dans cette dernière les moteurs se classent en raison inverse de leur complexité.
Le meilleur moteur est le plus simple, et la gamme possible en π et λ est assez faible :
$0,2 < \lambda < 0,8$ $10 < \pi < 14$

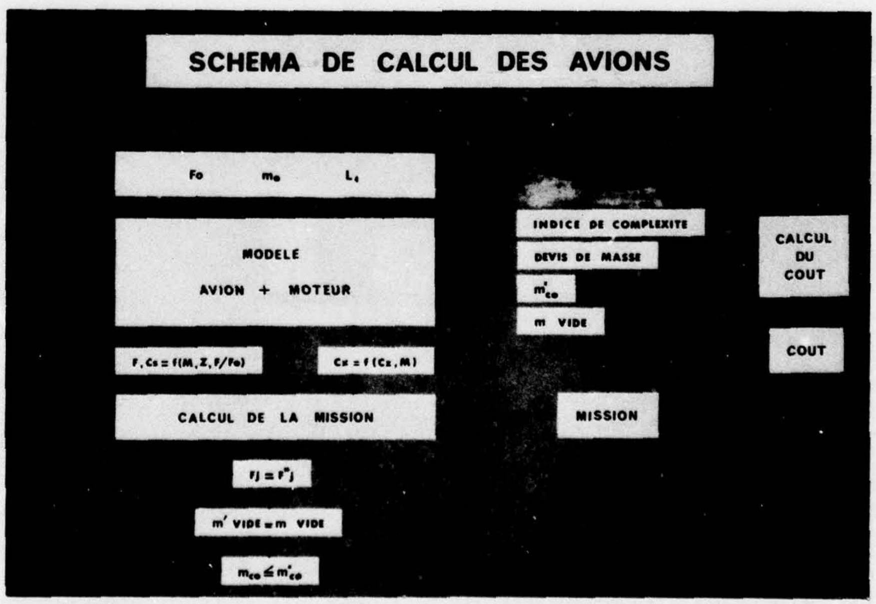
4) Conclusion.

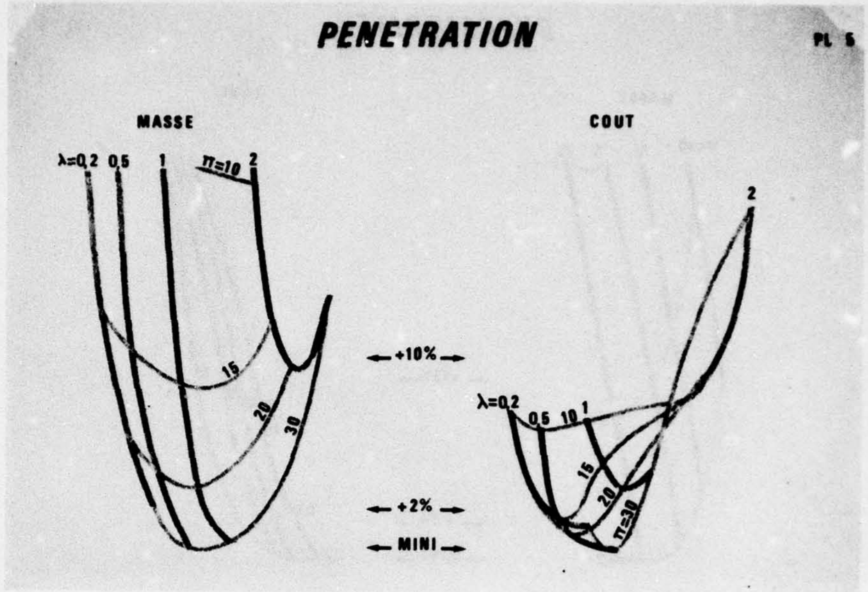
Le moteur moyennement dilué et très peu comprimé s'impose pour une mission d'interception difficile.

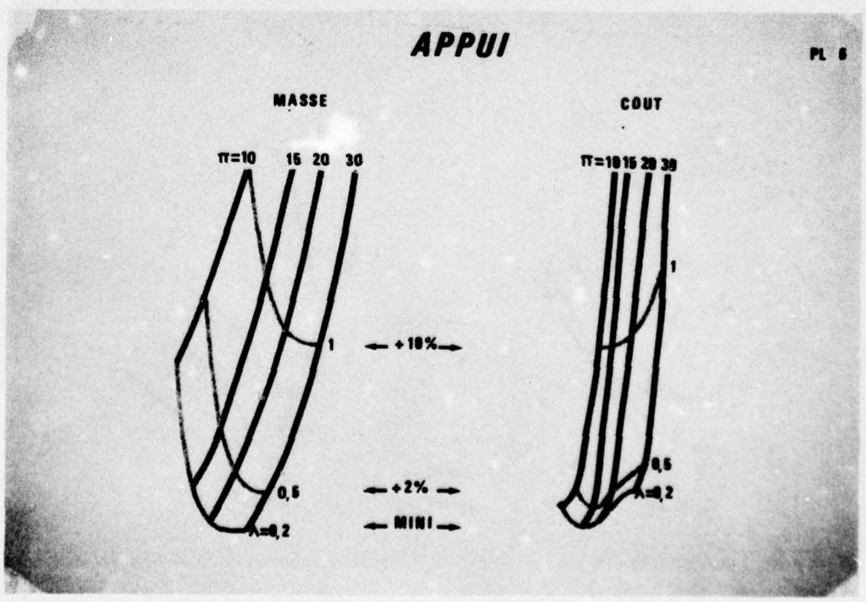

MODELE DE L'AVION
TAILLE
MISSION
MOTEUR
CAPACITE CARBURANT
ENTREE D'AIR
MASSE AU DECOLLAGE
MASSES FIXES
TYPE ADAPTATION
AVION
Cx = f(Cz,M)
CORRECTION MOTEUR
DEVIS DE MASSE
CAPACITE DE CARBURANT
m vide
PERFORMANCES
MASSES
COUT

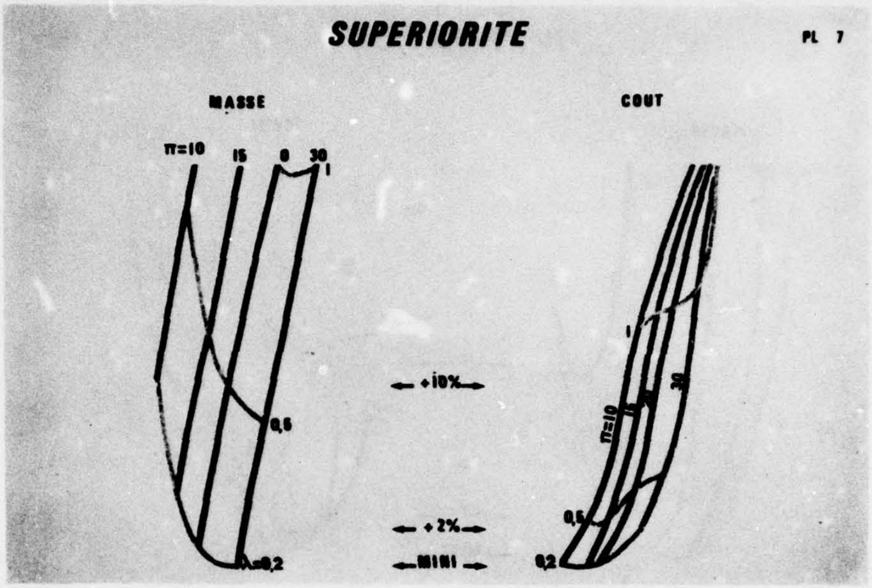

MODELE DU MOTEUR
AVION
CYCLE
TAILLE
CORRECTIONS ENTREE D'AIR
PC
moteur
MASSE
INDICE DE COMPLEXITE
PERFORMANCES
MASSES
COUT

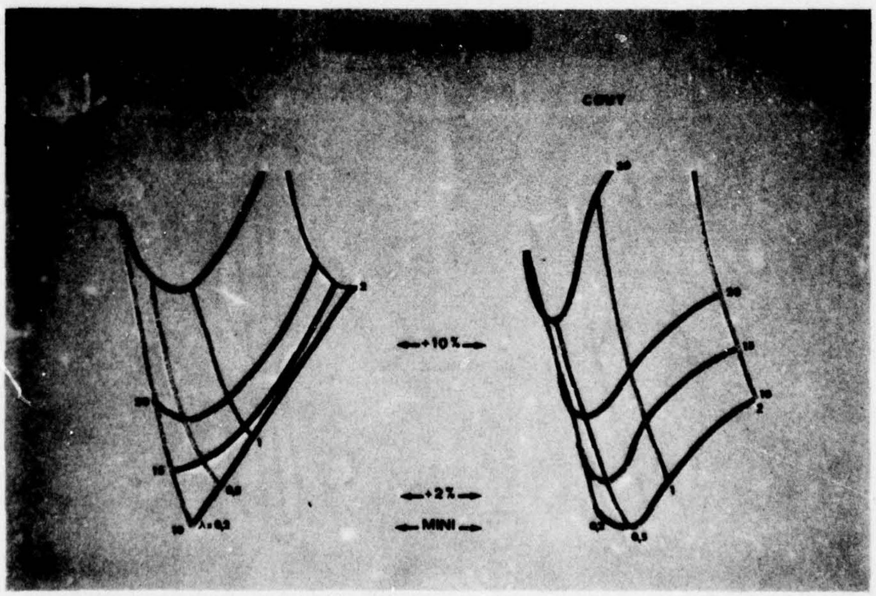

CONSOMMATION SPECIFIQUE
Z = 0
Z = 20 000 ft
SEC
PC
1,4
1,3
1,2
1,1
1,0
λ=0,5
1
2
π=10
15
20
30
3,0
2,5
2,0
π=10
15
20
30
2
1
λ=0,5


SCHEMA DE CALCUL DES AVIONS
Fo m_e L_t
MODELE
AVION + MOTEUR
F, Cs = f(M, Z, F/Fo)
Cx = f (Cz, M)
CALCUL DE LA MISSION
$r_j = r''_j$
m' VIDE = m VIDE
$m_{co} \leq m'_{co}$
INDICE DE COMPLEXITE
DEVIS DE MASSE
m'_{co}
m VIDE
CALCUL
DU
COUT
COUT
MISSION


PENETRATION
PL 5
MASSE
COUT
λ=0,2 0,5 1 π=10 2
15
20
30
+10%
+2%
MINI
λ=0,2 0,5 10 1
2
15
20
π=30


APPUI
PL 6
MASSE
COUT
π=10 15 20 30
1
0,5
λ=0,2
+10%
+2%
MINI
π=10 15 20 30
1
0,5
λ=0,2


SUPERIORITE
PL 7
MASSE
COUT
π=10
15
30
1
0,5
λ=0,2
+10%
+2%
MINI
0,2


+10%
+2%
MINI

IMPACT OF ACTIVE CONTROL ON STRUCTURES DESIGN

by

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INTRODUCTION

Active control technology (ACT) can have a large impact on structures design in terms of weight reductions for a given performance or improving performance with a given weight. For civil or military transport aircraft the measure by which it is usually judged is the direct operating cost. So the value of implementing ACT can be easily assessed for these airplanes. Many applications of ACT have been reported in this field starting with the famous LAMS program on the B 52 [1]. Further examples are the "B 52 Control Configured Vehicles Program" [2] the "C-5A Active Lift Distribution Control-System" [3] and an improvement of ride qualities of the Boeing 747 [4]. A complete review of these activities can be found in [5] and [6] which is more concerned with ACT applications on fighters. This paper deals only with the impact of ACT on fighter aircraft structures where the situation is much more complex due to the large variety of roles these airplanes have to fulfill compared with large transport aircraft.

It should also be indicated that a large part of this paper deals with structural dynamics and ACT which can be equally attributed to the fact that both authors are working in this field and that it is the most complicated application of ACT.

Three different motives for applying ACT can be recognized:

(i) Introduction of artificial stability leads to the control configured vehicle (CCV) and yields maximum gains only when this philosophy is adapted in the design stage.

(ii) Corrective action on an existing but unsatisfactory aircraft by using tools already available in modern fighters such as fly-by-wire electrohydraulically driven control surfaces, stability augmentation systems.

(iii) Improvement of already existing aircraft by exploiting its full control capacity. Ride quality improvement (for terrain following, high incidence buffet), weapon platform stabilization, gust alleviation, manoeuvre load control, weapon aiming, fatigue reduction, direct lift control, direct side force control, active flutter suppression for newly added stores would fall under this heading.

1. ACTIVE CONTROL TECHNOLOGIES

1.1 Artificial Lateral Stability

The fin area on fighter aircraft is determined by either the trim requirement for an engine failure at low q (twin engine fighters) the side wind landing requirement or the weathercock stability requirement. For the first case the concept of artificial lateral stability is not applicable. It is very likely that the fin size is defined by the stability requirement in the supersonic regime. In that case the fin area could be reduced by providing artificial lateral stability which in turn could reduce fin weight. If the original fin had to carry a weight penalty for flutter or gust load reasons then the weight would be further reduced.

1.2 Longitudinal Stability

If artificial stability is introduced then the c.g. can be located further aft and the full upward capability of the tailplane, or elevons can be utilized to trim in the high incidence condition. The download on the tail with flaps down is reduced with further aft location of the c.g.. Therefore the tailplane size may be reduced 30 % and usable lift increased by 15 %. A typical plot for tailplane sizing is shown in Fig. 1 taken from [7].

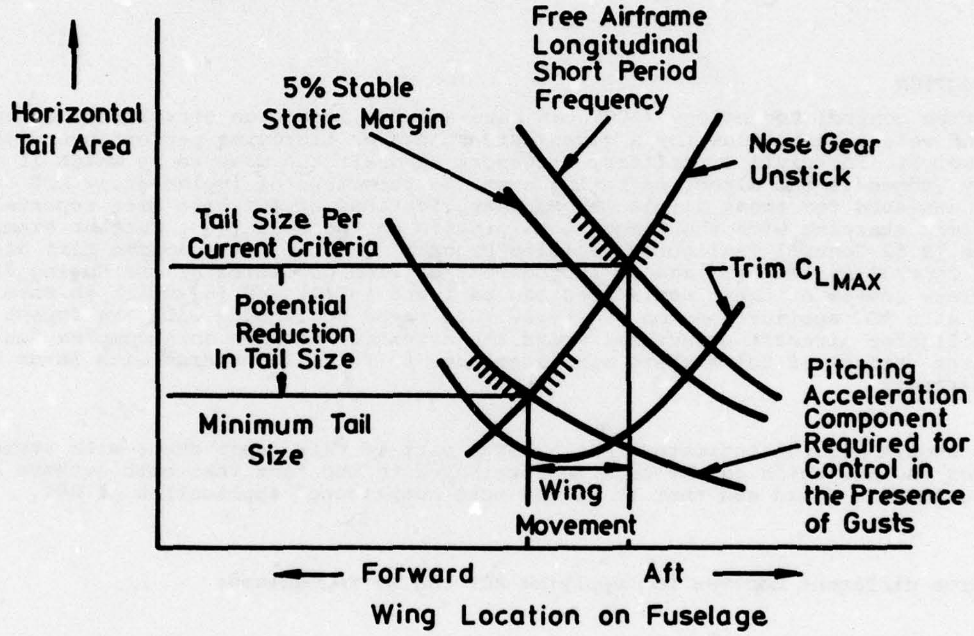


FIG. 1 TAILPLANE SIZING CRITERIA

Obviously the gain in usable lift is increasing with decreasing arm of the tailplane. This means an increase of usable lift by 30 % for tailless delta airplanes. Weight reductions of about 10 % have been demonstrated for a fighter A/C having artificial stability and equal performance as the conventional approach in Fig. 2,[8]. The amount of weight reduction is strongly dependent on the fixed weight fraction (weight which cannot be scaled down like crew, support system, avionics, specific preload). The smaller the fixed weight fraction, the greater is the gain due to use of artificial stability. Instead of reducing size and weight these gains can be transformed into higher performance but this is not the theme of this paper.

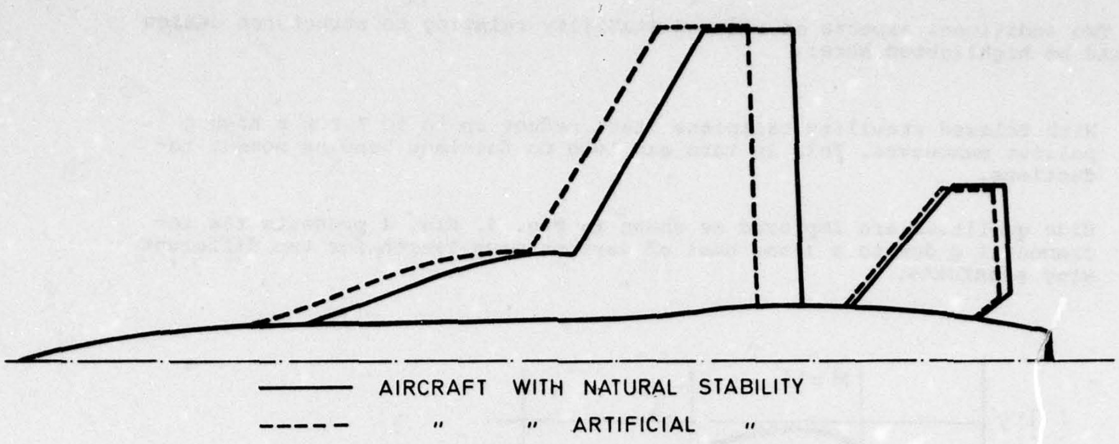


FIG. 2 AIR SUPERIORITY FIGHTER: COMPARISON AT SAME PERFORMANCE

It should be mentioned here that a tailplane flutter speed within the flight envelope (and the tailplane or fin are usually the most flutter critical surfaces) giving stiffness requirements or attachment (actuator) stiffness requirements or necessitating balance weights is increased with the reduction of area thus reducing tailplane weight again.

Fig. 3 shows the necessary mass balance weight as a fraction of total tailplane weight when the required speed envelope for a specific aircraft is cleared. Another, larger balance weight must be incorporated in the aircraft to reinstall the c.g. position for natural stability.

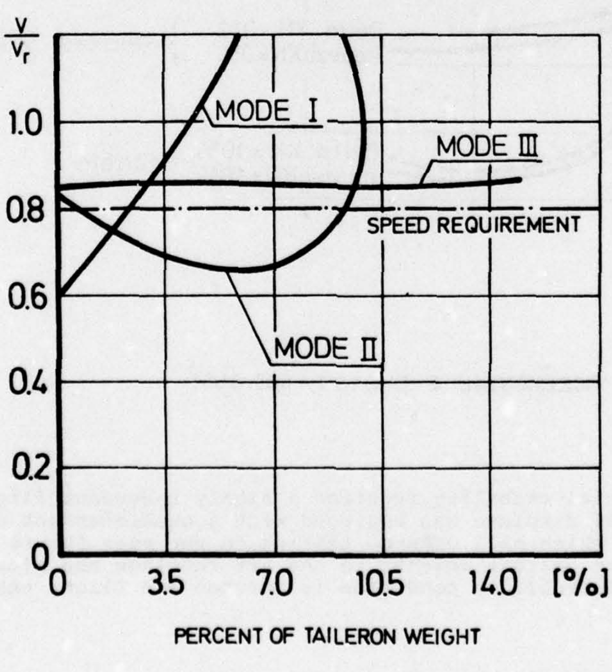


FIG. 3 FLUTTER SPEED VS. TAILPLANE MASS BALANCE

Two additional aspects of relaxed stability relating to structures design should be highlighted here:

a) With relaxed stability tailplane loads reduce up to 50 % for a high g pullout manoeuvre. This in turn can lead to fuselage bending moment reductions.

b) Ride qualities are improved as shown in Fig. 4. Fig. 4 presents the incremental g due to a 1-cos gust of various wave length for two different wing planforms.

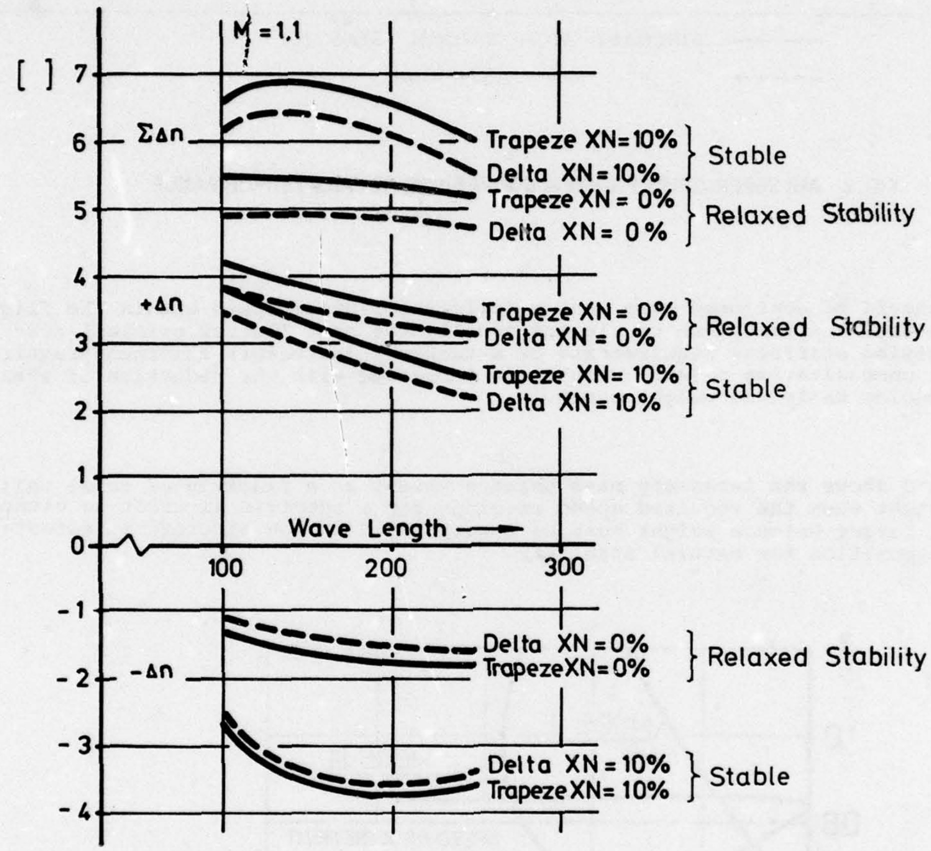


FIG. 4 INCREMENTAL G DUE TO 1 - COS. GUST

The concept of artificial stability requires a highly redundant flight control system. At MBB a F 104 airplane was equipped with a quadredundant digital flight control system, [9] which will undergo testing in the near future [10]. In several steps by putting ballast weights in the aft fuselage and attaching a canard the final relaxed stability condition is reached and flight tested.

The final chosen configuration is shown in Fig. 5.

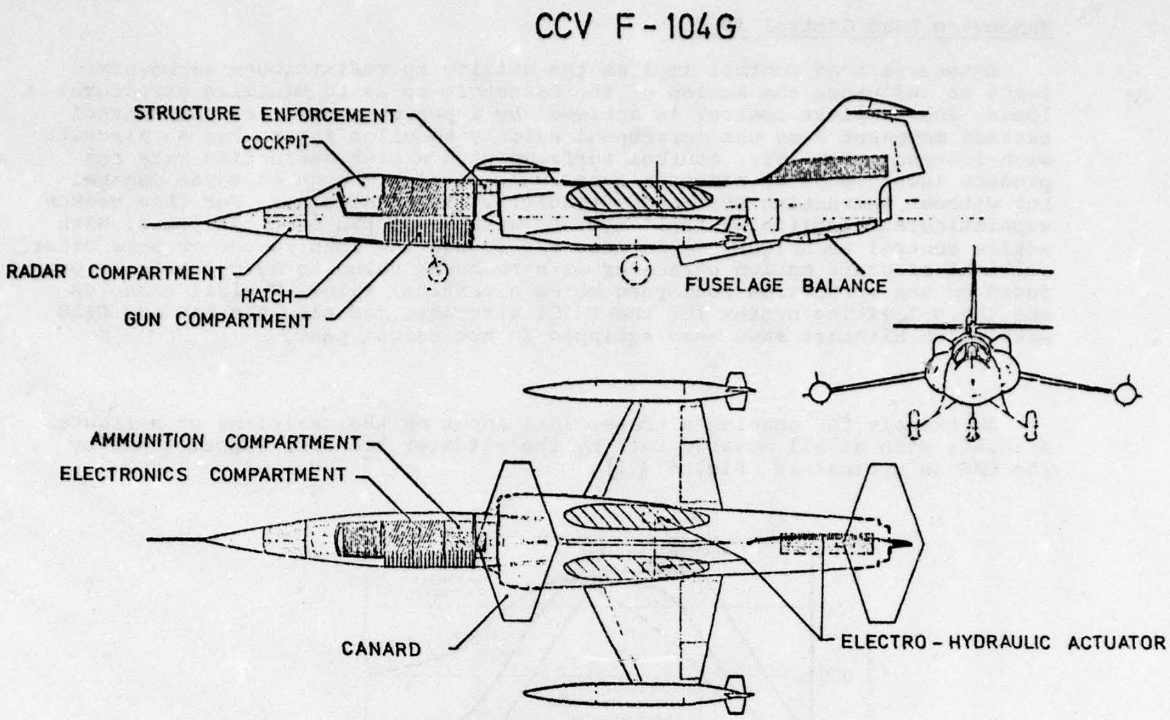


FIG. 5 F-104 G WITH ARTIFICIAL STABILITY

Before leaving the discussion of the concept of artificial stability a few words about the flight control system should be said. Having artificial stability for rigid body or elastic modes the failure of the control system would be fatal for the aircraft. In order to avoid this hazard the critical system components must be multiplied which in turn reduces the reliability of the total system. In [11] it is shown that the reliability of the hydraulic actuators is the dominating factor for flight safety. For this reason the complexity of surface actuators has increased during the last years, especially for fly-by-wire applications. Flight safety is improved by these multiple redundancies but the total reliability is decreased rapidely which is manifested in mission aborts and maintenance events.

These problems could be avoided if the control surfaces are multiplied and the summation of control moments is done aerodynamically. The disadvantage of this method is, that the available control moments are reduced when surface actuators are failing. This disadvantage is only important for low q because efficiency increases with q. The overruling consideration is that surface actuators can be made simplex.

If such a design is chosen, then there is a wide field opened for application of ACT. Because now it is possible to use selected surfaces for various ACT approaches not requiring high reliabilities.

1.3 Manoeuvre Load Control (MLC)

Manoeuvre Load Control implies the ability to redistribute aerodynamic loads or influence the motion of the manoeuvre so as to minimize structural loads. The simplest control is achieved by input shaping, i.e. the control surface movement does not correspond exactly to pilot input. For an aircraft with inherent stability, control surfaces with a high deflection rate can produce local loads on aircraft substructures high enough to cause damage, but without increasing the manoeuvreability of the aircraft. For this reason sophisticated "artificial feel" systems were developed over the years. With active control techniques it is possible to use the load factor or some other critical aircraft motion parameter as a feedback value to keep the loads produced on the structures concerned below a critical value. Typical examples are the g-limiting system for the F-104 aircraft, and also the SAS and CSAS with which aircraft have been equipped in the recent past.

An example for shaping a trapezoidal input on the tailplane of a fighter aircraft with an all movable tail by the actuator transfer function and by the CAS is present in Fig. 6 [12].

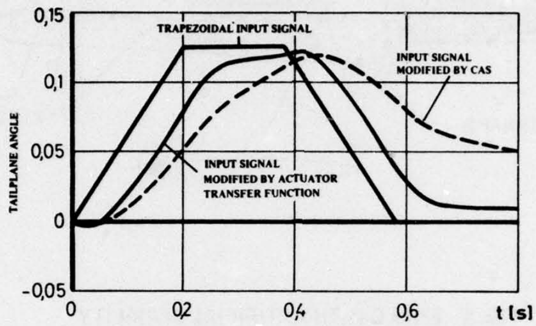


FIG. 6 TAILPLANE ANGLE INPUT SIGNAL

The tailplane loads are shown in Fig. 7 for the input signals of Fig. 6. A reduction of these loads of about 30 % can be seen when the full CSAS is engaged. It is also presented in this figure that there is little difference (for the fighter investigated!) between the results with rigid and elastic aircraft.

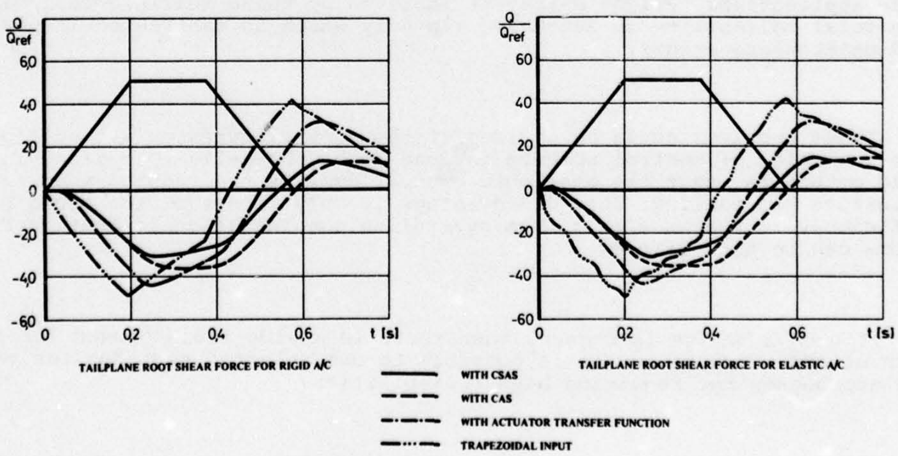


FIG. 7 TAILPLANE ROOT SHEAR FORCE DUE TO MANEOUVRE

Another application for MLC is decreasing dynamic incremental loads induced by mass cross coupling with the aid of secondary control surface movements, an example of which will be shown for a fighter aircraft for the roll manoeuvre as a function of the load factor at the beginning of the manoeuvre [13], [14]. The initial load factor is caused by straight and level flight or by curved flight paths. The primary control for the rolling manoeuvre is the aileron; the secondary control is the elevator or rudder. The aim is to keep the deviation of the load factor from the initial value as small as possible. The increase in the load factor is caused by the pitch motion of the aircraft during roll. The figures show the minimum and maximum load factors occuring during a roll as a function of the initial load factor.

Fig. 8 shows the influence of a rudder deflection geared to the aileron. Fig. 9 shows the influence of an additional elevator deflection on the additional load factor caused by the roll, the elevator deflection being a function of the load factor. The dotted curves delineate the region where the rolling manoeuvre is flown solely with the aileron, and the solid curves, where a secondary control surface was used in additon to the aileron. The latter manoeuvre, where the elevator deflection was kept dependent on the load factor, resulted in the smallest deviation from the initial load factor.

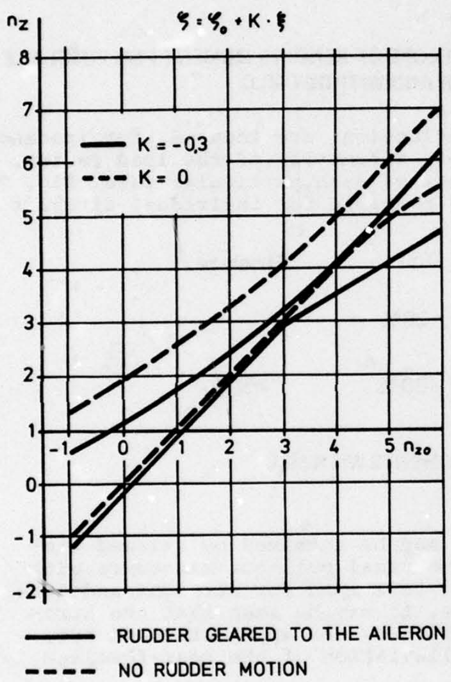


FIG. 8 ROLL MANOEUVRE IMPROVED BY AN ADDITIONAL RUDDER DEFLECTION

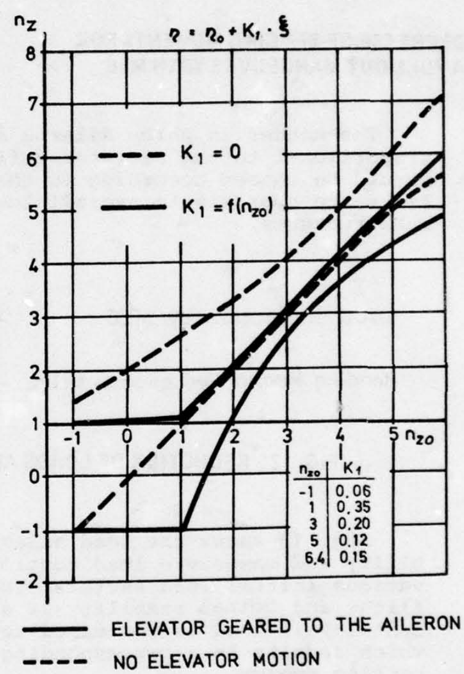


FIG. 9 ROLL MANOEUVRE IMPROVED BY AN ADDITIONAL ELEVATOR DEFLECTION

As a typical example for the application of MLC the redistribution of loads is shown here for a symmetric pull-out manoeuvre of a fighter aircraft with a high initial pull-out load factor. Such a manoeuvre can be initiated by an elevator deflection only, or by up-elevator plus symmetrical up-aileron, or by up-elevator, symmetrical up-aileron and symmetrical flaps down.

The time histories of the deflections and load factors during the manoeuvre is virtually the same for all three modes of actuation, whereas the aerodynamic load distributions differ for the wing and the horizontal tail. When the ailerons and flaps are actuated simultaneously for the pull-out manoeuvre, aerodynamic wing loads increase on the inboard wing, decrease at the tip, and decrease over the entire horizontal tail. This leads to a decrease in the bending moments on the wing, the fuselage, and the horizontal tail, as shown in Fig. 10. The dotted lines show bending moments without MLC, the solid lines those with MLC. Fig. 11 shows, for the same manoeuvre, the envelopes of the fuselage bending moments with and without MLC.

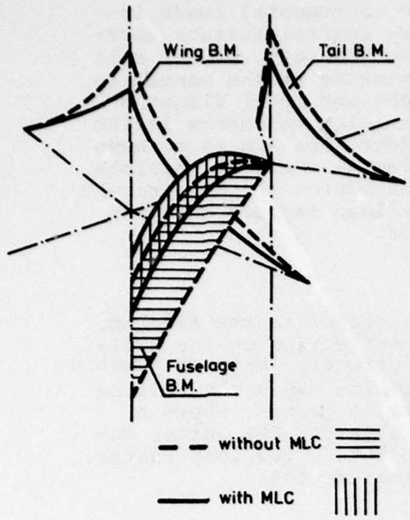


FIG. 10 DECREASE OF BENDING MOMENTS FOR A PULL-OUT MANOEUVRE WITH MLC

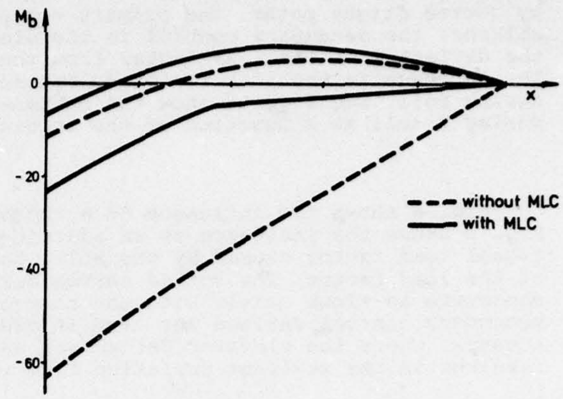


FIG. 11 ENVELOPE OF BENDING MOMENT FOR FUSELAGE WITH AND WITHOUT MLC

The manner in which aileron and flaps deflections are induced, for instance proportional to the elevator deflection, or as a function of the load factor, should be chosen according to the requirements of each particular case. Fig. 12 lists the approximate overall load decreases possible for individual aircraft substructures.

	Wing	Tail	Fuselage
Load Reduction by MLC :	~8 - 10%	~20%	
Bending Moment Reduction by MLC:	~13%	~20%	~50%

FIG. 12 REDUCTION OF LOADS AND BENDING MOMENTS WITH MLC

Fig. 13 shows the load alleviation that may be attained by relaxed stability and manoeuvre load control for a symmetrical pull-out manoeuvre with various initial load factors. The horizontal-tail load for straight and level flight and normal stability is set to 1 here. It may be seen that the horizontal-tail load is decreased to almost a third by the application of ACT, which results in a correspondingly higher alleviation of the rear-fuselage bending moment.

	Q/Q_{Ref}		
	Inherent Stability	Relaxed Stability	Manoeuvre Load Control
$n_z = 1$	1.0	0,5	0,5
$n_z = 8$ (Pull-Out Manoeuvre)	7,0	3,5	2,5

$Q_{Ref} = Q$ at $n_z = 1$

FIG. 13 HORIZONTAL TAIL LOAD Q/Q_{REF} FOR RELAXED STABILITY AND MLC

The manoeuvrability of a fighter aircraft may be improved with the aid of leading and trailing edge flaps which are operated as a function of angle of attack, total pressure, and load factor, because then manoeuvres can be flown optimally at high lift and low drag. The resulting in profile drag is partly balanced by the reduction in induced drag provided by the flow acceleration effect of the leading edge flaps.

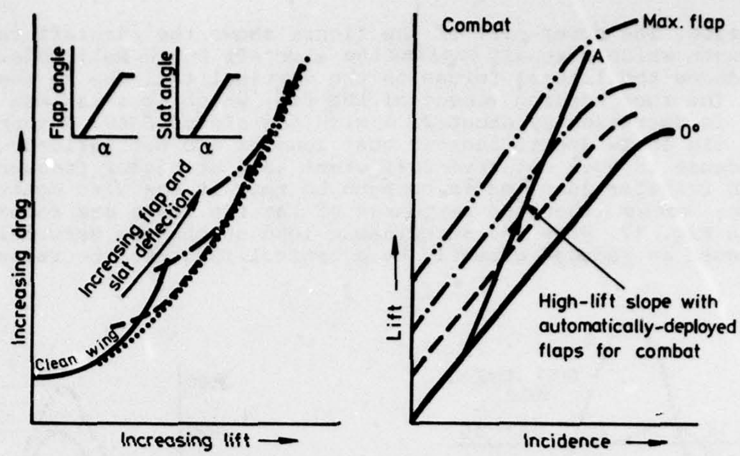


FIG. 14 INCREASE OF MANOEUVRABILITY WITH COMBAT FLAPS

Fig. 14 shows qualitatively the drag envelope (dotted line) as a function of the deflections of the high-lift devices [15]. The right-hand part shows the lift as a function of angle of attack, taking into account the high-lift devices and their high lift-coefficients essential for combat manoeuvres. The wing root bending moment of the wing with extended combat flaps is smaller than that of the clean wing, because the combat flaps redistribute the aerodynamic loads in such a way to reduce the wing bending moment. These examples showed how load distributions can be altered and manoeuvreability can be increased, with the aid of ACT and "manoeuvre load control", leading to weight savings or performance improvements, or both. Active Control Techniques offer a wide field of action for improvements here.

1.4 Gust Load Alleviation

The additional load factor which an aircraft experiences in turbulent air (see Fig. 15) is the higher, the higher its lift coefficient, and the lower its wing loading. The capabilities provided by optimally controlled high-lift devices for combat manoeuvres shown in Fig. 14 turn a combat aircraft into a strongly gust-sensitive vehicle, during a "terrain-following" mission since these high-lift devices raise the lift coefficient and reduce wing loading. This gust sensitivity can be decreased considerably during a "terrain following" mission by solely using trailing edge flaps at small angles of attack, whose motion is controlled by measured gust velocity.

$$\Delta n = \frac{\eta \cdot U_{B\ddot{o}} \cdot V \cdot C'_\alpha}{16 \frac{G}{F}}$$

$U_{B\ddot{o}}$ = GUST VELOCITY

V = FLIGHT SPEED

$\frac{G}{F}$ = WING LOADING

C'_α = LIFT DERIVATIVE

η = ALLEVIATION FACTOR

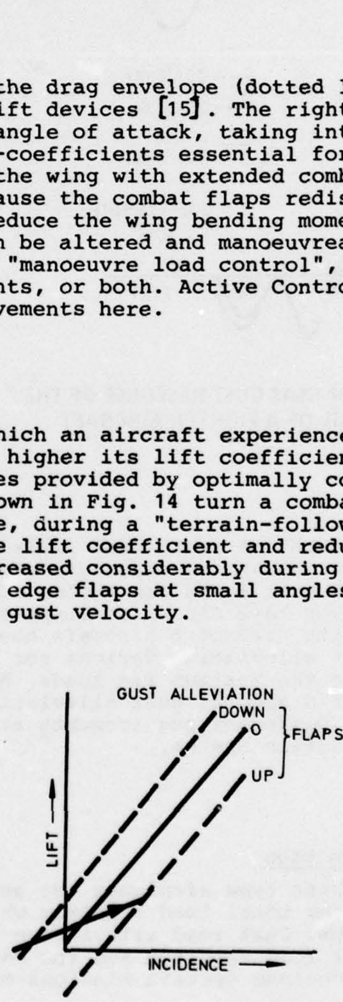


FIG. 15 IMPROVEMENTS FOR GUST SENSITIVITY BY FLAPS

This leads to the topic of "gust alleviation", i.e. the technique of suppressing the rigid-body response of the aircraft to gust excitation. For this purpose measured gust velocities or response quantities of the aircraft or its substructures are used to control flap and control surfaces to minimize aircraft loads. Fig. 16 shows the response of a combat aircraft equipped with CSAS to a solitary gust of the shape (1-cos) by depicting fin loads. The CSAS acts on taileron and rudder, but was not designed for gust

alleviation. The upper part of the figure shows the aircraft response to a gust length which strongly excites the aircraft Dutch Roll mode. Here the CSAS reduces the lateral forces on the vertical tail due to the aircraft motion. The root bending moment of the fin, which is a maximum for this gust length, is decreased by about 25 % with the aid of CSAS. Responses in the form of fin loads due to shorter gust lengths are not influenced by the CSAS because it does not have sufficient gain at higher frequencies. If the CSAS transfer function is changed to that it can also control higher frequency modes then the responses of the fin loads are reduced too, as shown in Fig. 17. Here the aerodynamic load on the the vertical tail caused by the gust is reduced directly by a control input to the rudder.

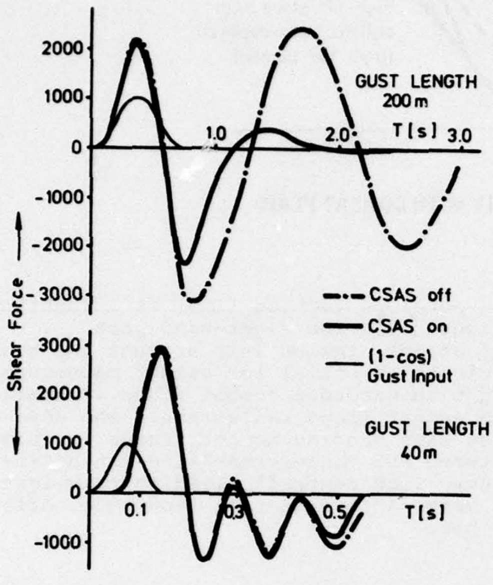


FIG. 16 INFLUENCE OF CSAS GUST RESPONSE OF THE VERTICAL TAIL OF A FIGHTER AIRCRAFT

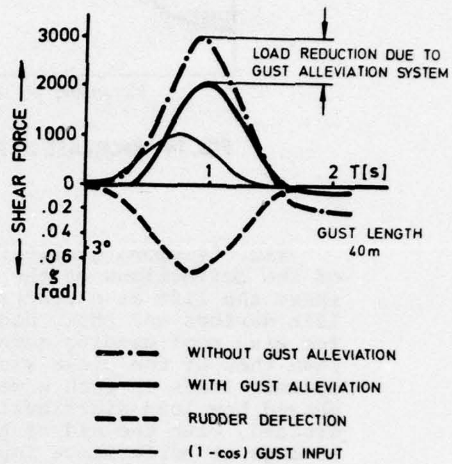


FIG. 17 INFLUENCE OF A "GUST LOAD ALLEVIATION SYSTEM" IN VERTICAL TAIL LOADS

For a combat aircraft gust loads on the wing and horizontal tail are not critical, because both components are designed for high manoeuvre loads. Thus only lateral gust loads can become critical for the fin. Modern combat aircraft, however have SAS or at least a yaw damper. Experience with combat aircraft and with transport aircraft have shown that SAS or yaw dampers act automatically as gust alleviation devices for large gust lengths. The latter ones generally cause the maximum fin loads. By suitable design of the SAS or yaw damper the need for a special gust alleviation system for a combat aircraft may be eliminated. Different requirements arise, however, in connection with a Fatigue Reduction System.

1.5 Fatigue Reduction

For fighter type airplanes the gust spectrum is giving only small contributions to the total load spectrum which is formed mainly by the manoeuvre load spectrum. Gust load alleviation systems therefore have little influence on the total load spectrum and the manoeuvres which would reduce it cannot be limited because certain missions must be fulfilled.

The possibilities which have been shown for manoeuvre load control leading to a reduction of loads and bending moments such as

a) load redistribution on a total airplane
b) reduction of dynamical effects (such as mass cross couplings)
c) shaping of control surface motions

also reduce the manoeuvre load spectra and therefore lead to a fatigue reduction. In Fig. 18, [16] the effect of a MLC redistributing the wing loading is shown.

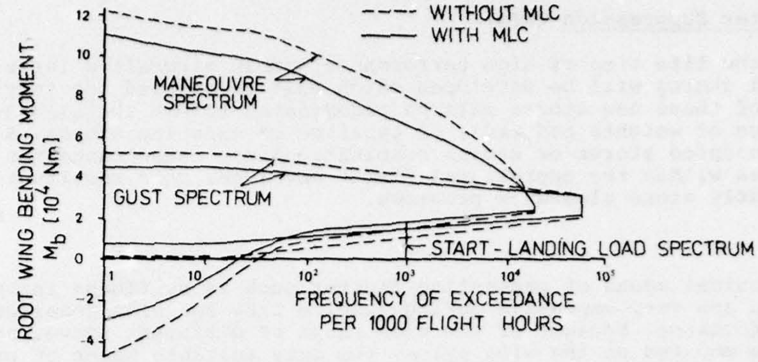


FIG. 18 GUST AND MANEOUVRE LOAD SPECTRA OF A COMBAT AIRCRAFT

1.6 Ride Quality Improvement

Fighter airplanes relying on low level flying at high speed to navigate or to avoid radar screens are exposed to high atmospheric turbulence. The arising accelerations can have a severe detrimental impact on crew performance. Variable wing sweep is an obvious solution to reduce gust sensitivity and still have excellent manoeuvre capability. ACT also offers the possibility to improve ride qualities by reducing accelerations with the controlled moving of surfaces. If it is possible to introduce the forces and measure the motions at the place where they shall be reduced (ILAF-System) then this is technically preferably because there is no phase shift between the force to be introduced and the measured accelerations. Such a system was chosen for the B1 bomber with canards at the crew compartment. Sometimes the accelerations are arising from weakly damped fuselage modes which can be influenced by ACT if fast responding actuators are available. Fig. 19 shows the impact of a gust alleviation system on ride qualities.

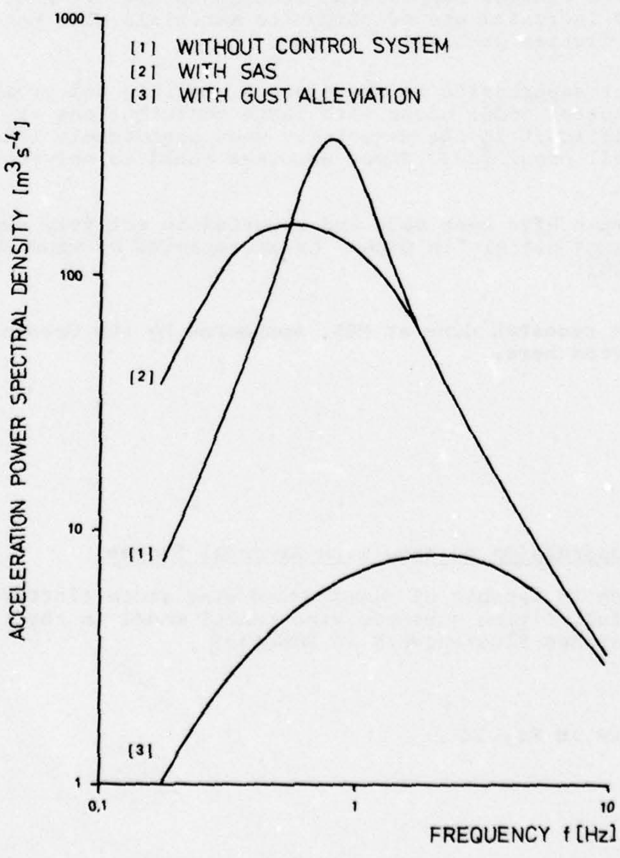


FIG. 19 POWER SPECTRUM OF THE PILOT'S SEAT ACCELERATION

1.7 Active Flutter Suppression (AFS)

During the life time of high performance combat aircraft a large number of new external stores will be developed which must be carried for tactical reasons. Most of these new stores will be accomodated within the already flutter cleared range of weights and radii of gyration of existing stores. Some of these wing mounted stores or stores combinations may cause dangerous flutter instabilities within the operational flight envelope. This results in extensive and costly store clearance programs.

The classical means of preventing flutter such as stiffness increase or mass balance are very expensive during service life and also penalize the aircraft's performance. Because of the wide range of different stores and store combinations mounted on the wing pylon, the only suitable means of preventing flutter is limiting the speed. This in turn could mean an increase in the vulnerability of the aircraft during a ground attack mission.

Many contemporary fighter attack aircraft have flutter restrictions in the range of Ma = 0.7 to Ma = 1.0 at sea level. For large and unconventional stores flutter occurs even at Ma = 0.4 S.L.

In some cases (especially asymmetric store combinations) flutter was not actually encountered; low damping, however, limited the performance of the aircraft.

The inventory of advanced fighter-attack aircraft of the future will most likely be limited in system type and quality because of soaring development costs. This trend will dictate a high performance combat aircraft and the use of many external store configurations. Based on the growth patterns of aircraft/weapon systems of the past, store flutter limits on these advanced aircraft appear inevitable and within the same speed range as they are found today unless improved flutter suppression techniques are defined. It is also believed that the increased use of composite materials will not solve all these wing store flutter problems.

Active flutter suppression is therefore a possible and promising solution. Sometimes also flutter modes occur with large contributions of fuselage motions which are very difficult to use especially when aerodynamic interferences between wing and tail occur [29]. These problems could be solved with AFS.

Several attempts have been made and reported to actively suppress flutter on fighter airplanes either "on paper" or accompanied by wind tunnel tests [17], [18], [19], [20].

As an example research done at MBB, sponsored by the German Ministry of Defense, is reported here.

1.7.1 Active Flutter Suppression on Wing with External Stores

A system which is capable of suppressing wing store flutter was developed and tested on a full flying subsonic wind tunnel model in the flutter tunnel of the Eidgenössisches Flugzeugwerk in Emmen [21].

The model is shown in Fig. 20 .

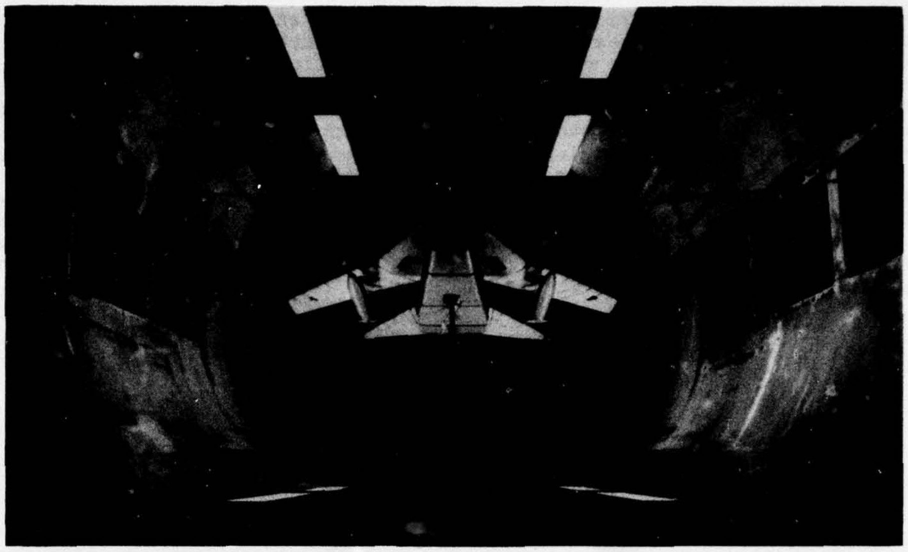


FIG. 20 SUBSONIC FLUTTER MODEL

Fig. 21 shows a block diagram of the vane control system.

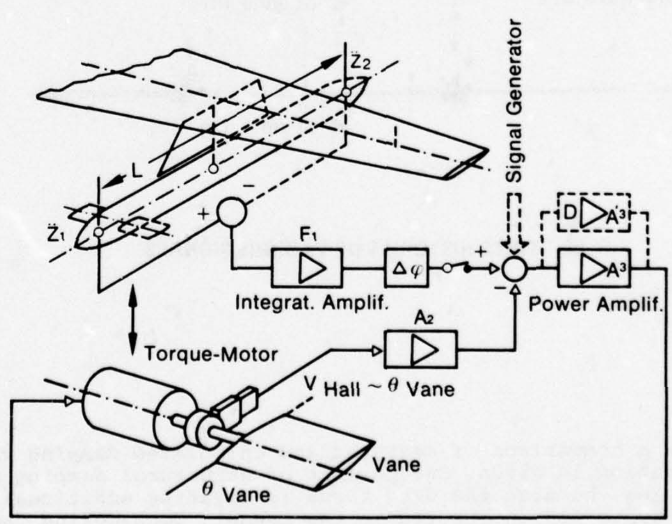


FIG. 21 BLOCK DIAGRAM OF THE VANE CONTROL SYSTEM

The control system drives a vane attached to a store controlled by a feed back signal in such a way that it counteracts the store motion. The developed mechanism can also be utilized for conventional flight flutter testing excitation techniques such as frequency sweep or harmonic signals together with a method for a quick finding of the frequency and damping of the flutter mode. Another application of the system is the reduction of external store amplitudes created by buffet or air turbulence, thus increasing the fatigue life of wing pylon attachments, improving the target aiming of weapons and sharpness of pictures shot by reconnaissance cameras in wing mounted pods. Fig. 22 shows the time histories for speeds above the flutter speed. These graphs were produced in the wind tunnel by switching the active flutter suppression system (AFSS) on and off. Since the system is unstable, it flutters when the suppression is off and it decays when it is on.

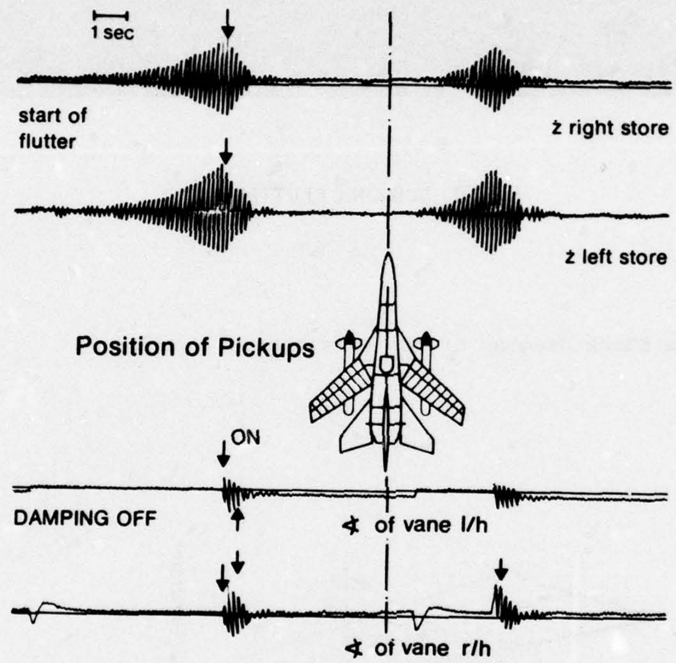


FIG. 22 TIMES HISTORIES OF VARIOUS SIGNALS

In Fig. 23 a comparison of measured and calculated damping values for a store configuration is given. One percent of structural damping must be added to mode 2 damping, because the drag force is creating additional damping when the model was supported on its rod in the tunnel. Considering the complicated flutter mechanism correlation of test and analysis is very good. The analysis underestimates the tunnel flutter speed only by about ten percent (AFSS off) and gives the same damping trend (AFSS off and on).

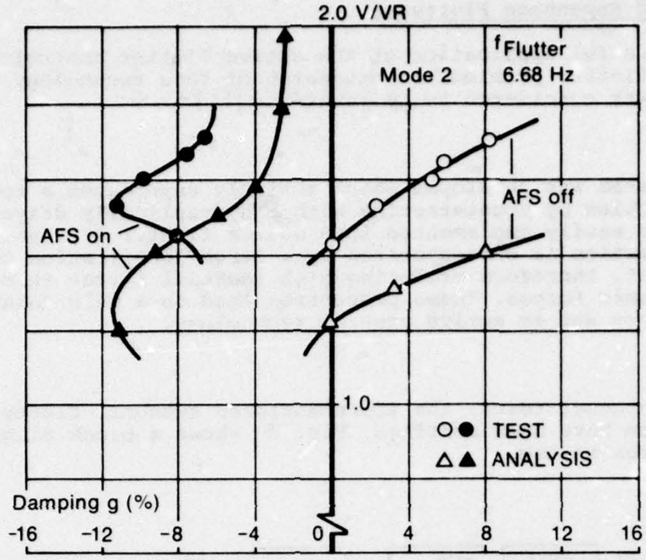


FIG. 23 COMPARISON OF MEASURED AND CALCULATED VELOCITY

Because the model is free flying in the tunnel a high angle of attack could be simulated that caused wing stall and a high noise environment.

Fig. 24 shows that the AFSS reduces the response of the store considerably. The wing response is not attenuated as much. This is due to the fact that the wing mainly responds in its bending mode. Very little damping force can be introduced into this mode at the wing pylon station.

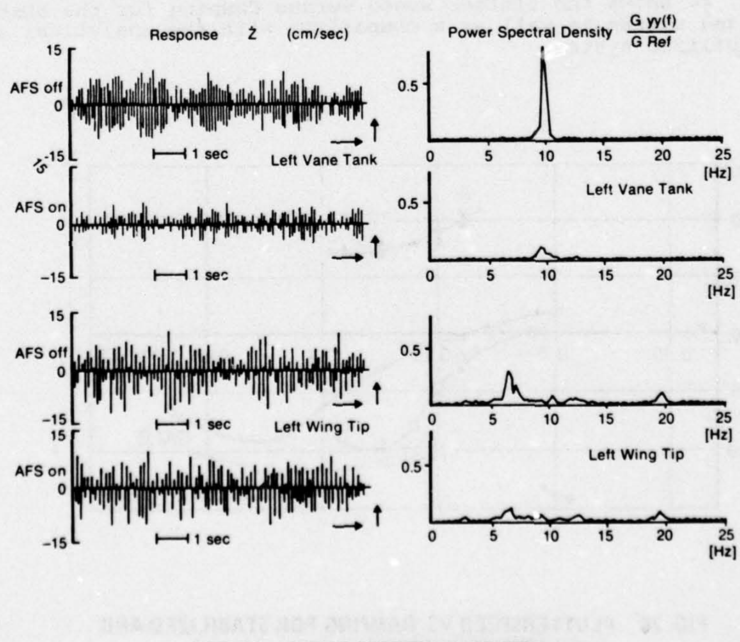


FIG. 24 MODE BUFFET RESPONSE

1.7.2 Active Control of Empennage Flutter

After a successful application of the active flutter control technology on a wing store flutter problem, an extension of this technology to an empennage flutter problem was considered to be rewarding [22].

A control system was developed which actively suppresses a total airplane model flutter problem by counteracting with a hydraulically driven rudder. Such a system could be easily implemented into modern fighter designs. The flutter phenomenon suppression is characterized by a large contribution of fuselage torsional movement, therefore producing high inertial forces in comparison with unsteady aerodynamic forces. These properties lead to a mild onset of flutter. Such a mode is very apt to active control technology.

For the wind tunnel tests, the aforementioned subsonic flutter model and the control system have been modified. Fig. 25 shows a block diagram of this flutter suppression system.

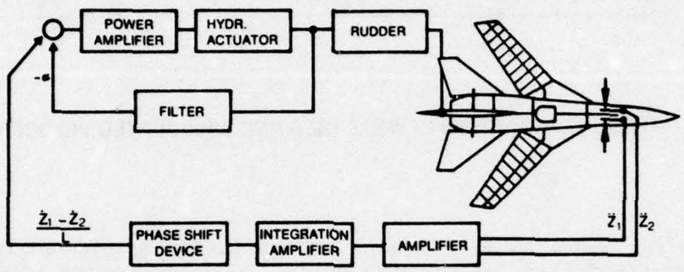


FIG. 25 SYSTEM BLOCK DIAGRAM

It can be said that the aim to control an empennage flutter mode was achieved. Fig. 26 shows the flutter speed versus damping for the stabilized and unstabilized system as well as a comparison with the analytical prediction for the unstabilized system.

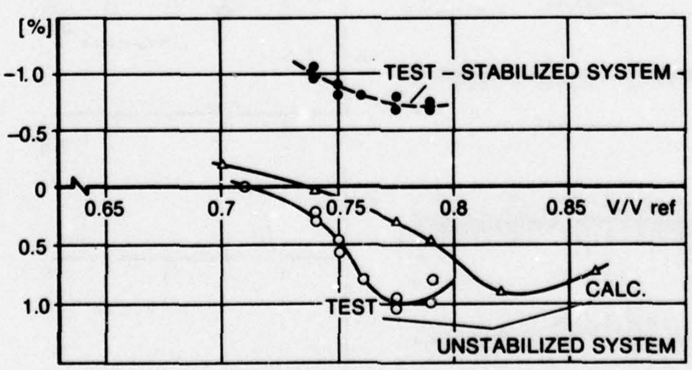


FIG. 26 FLUTTERSPEED VS. DAMPING FOR STABILIZED AND UNSTABILIZED SYSTEMS

1.7.3 Flight Test Program of a Wing Store Flutter Suppression on FIAT G 91

A follow-up effort was focused on the flight test of a wing/store flutter suppression system with store mounted vanes [23] . This program, started in 1975, with the design of the system and the instrumentation of a FIAT G 91 as flying test bed.

Test objectives of this study were to:

- Provide first flight experience with a flutter suppression system on external stores.
- Evaluate new methods of flight flutter testing of external stores flutter.
- Investigate the system's behaviour in turbulent air.

The aircraft with stores and AFSS is shown in Fig. 27 .

FIG. 27 FIAT G91 WITH AUTOMATIC FLUTTER SUPPRESSION SYSTEM

The aim of flutter suppression could be demonstrated in flight and the automatic mode excitation was successfully applied. Time histories of the stable and artificially unstable system are shown in Fig. 28 .

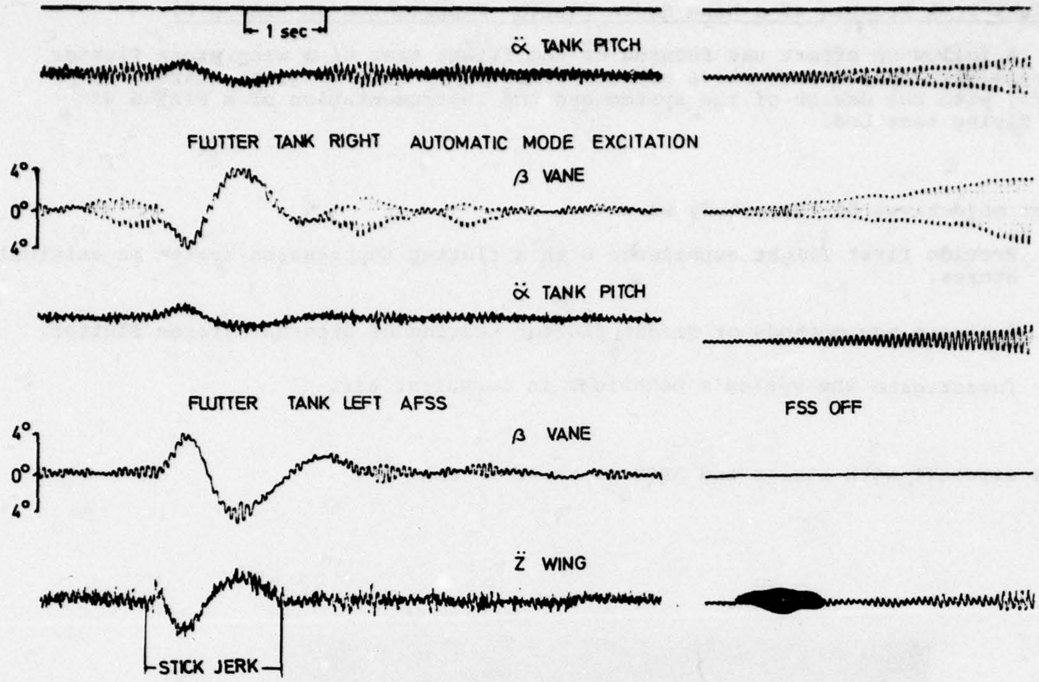


FIG. 28 TIME HISTORIES OF STABLE AND ARTIFICIAL UNSTABLE SYSTEM

The efficiency of the system is presented in Fig. 29.

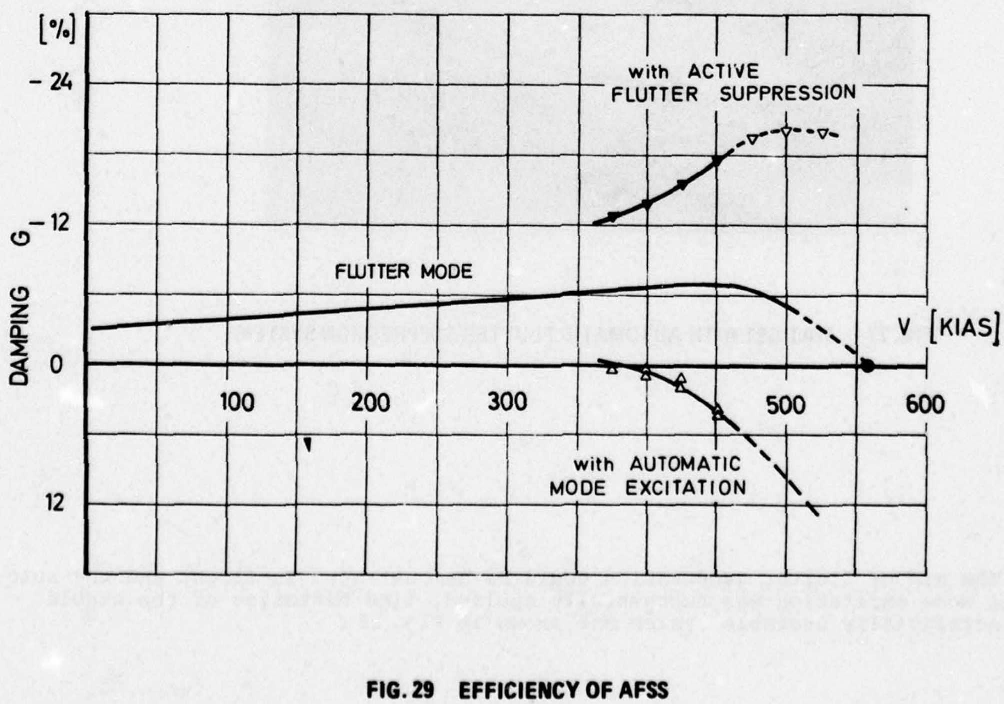


FIG. 29 EFFICIENCY OF AFSS

1.7.4 Active Flutter Suppression on a F4 Phantom Aircraft

A flight test program will be performed to study flutter suppression using already existing control surfaces (ailerons) and sensors on the wing of the F-4F. This program will be conducted in cooperation with the AFFDL (Air Force Flight Dynamics Laboratory). The objective of this program is to develop and flight test an active flutter suppression system (AFSS) which can become a possible candidate for an operational flutter suppression system on any aircraft. A flutter calculation for the proposed system is shown in Fig 30 .

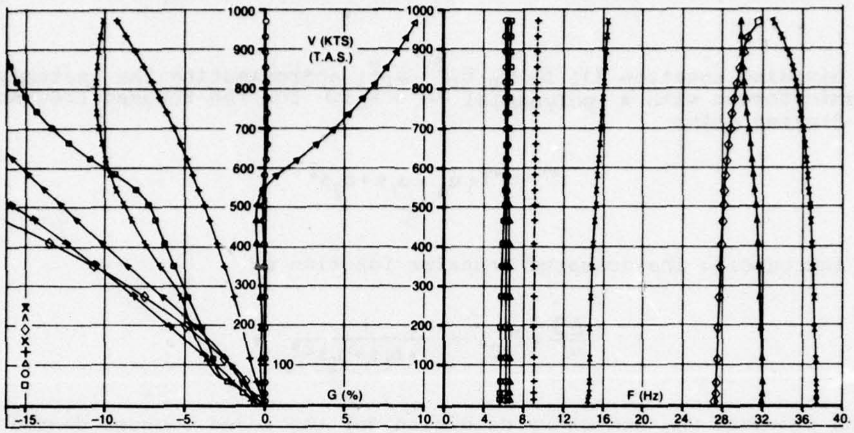


FIG. 30 DAMPING AND FREQUENCY VS. SPEED FOR F-4F WITH EXTERNAL STORES

1.7.4.1 Use of Optimal Control Laws for Flutter Suppression

Theoretical applications of optimal control laws for control of an elastic aircraft have been reported in [24] and [25] . The proposed technical approach for the F-4F program will be described here.

The equations of motion for the forced dynamic response of an aeroelastic system can be written in matrix differential equation form:

$$m_r b_r^2 \begin{bmatrix} M_{qq} & M_{q\beta_0} \\ M_{\beta_0 q} & M_{\beta_0\beta_0} \end{bmatrix} \begin{Bmatrix} \ddot{q} \\ \ddot{\beta}_0 \end{Bmatrix} + \frac{s_R}{k\,v} \left\{ \left[\omega_r^2 m_r b_r^2 \begin{vmatrix} gK_{qq} & 0 \\ 0 & K''_{\beta_0\beta_0} \end{vmatrix} + \frac{\rho}{2} v^2 F s_R \frac{b_r^2}{s_R^2} \begin{vmatrix} C''_{qq} & C''_{q\beta_0} \\ C''_{\beta_0 q} & C''_{\beta_0\beta_0} \end{vmatrix} \right] \right\} \begin{Bmatrix} \dot{q} \\ \dot{\beta}_0 \end{Bmatrix}$$

$$+ \left\{ \left[\omega_r^2 m_r b_r^2 \begin{vmatrix} K_{qq} & 0 \\ 0 & K'_{\beta_0\beta_0} \end{vmatrix} + \frac{\rho}{2} v^2 F s_R \frac{b_r^2}{s_R^2} \begin{vmatrix} C'_{qq} & C'_{q\beta_0} \\ C'_{\beta_0 q} & C'_{\beta_0\beta_0} \end{vmatrix} \right] \right\} \begin{Bmatrix} q \\ \beta_0 \end{Bmatrix} = \{Q(t)\} \qquad (1)$$

where m, b_r and ω_r are chosen reference mass, length and frequency and M, K and C are referred to as the generalized mass, stiffness and aerodynamic matrices which are nondimensional. The true airspeed v and the semispan S_R of the reference plane are used to form the reduced frequency $k = (\omega s_R)/v$. F is the area of reference plane and g is the structural damping of the elastic modes. The generalized forces Q are equal to zero for the conventional flutter problem. The generalized coordinate q describes the amplitude of the rigid body modes and the elastic airplane modes including elastic control surface modes for a system with actuators assumed to be rigid, whereas β_0 denotes the rotation of the rigid control surface according to the complex actuator stiffness represented by the impedance function of equation (2).

$$K_{\beta_0\beta_0} = K'_{\beta_0\beta_0} + iK''_{\beta_0\beta_0} \qquad (2)$$

For the controlled aircraft the servo-induced control deflection $\Delta\beta$ has to be introduced as an additional degree of freedom for each control surface. The generalized forces Q generated by the servo-induced control deflections $\Delta\beta$ can be described as the right-hand term of equation (1) by

$$\{Q(t)\} = -m_r b_r^2 \begin{Bmatrix} M_{q\Delta\beta} \\ M_{\beta_0\Delta\beta} \end{Bmatrix} \Delta\ddot{\beta} - \frac{\rho}{2} v^2 F s_R \frac{b_r^2}{s_R^2} \frac{s_R}{k \cdot v} \begin{Bmatrix} C''_{q\Delta\beta} \\ C''_{\beta_0\Delta\beta} \end{Bmatrix} \Delta\dot{\beta} - \frac{\rho}{2} v^2 F s_R \frac{b_r^2}{s_R^2} \begin{Bmatrix} C'_{q\Delta\beta} \\ C'_{\beta_0\Delta\beta} \end{Bmatrix} \Delta\beta \qquad (3)$$

Assuming normalized rigid control surface modes β_0 and $\Delta\beta$, the rotation of each control surface can be superimposed by

$$\beta = \beta_0 + \Delta\beta \tag{4}$$

For the case described here β is the aileron rotation.

Dividing equation (1) by $m_r \cdot b_r^2 \cdot \omega_r^2$, approximating the unsteady aerodynamic forces with a polynomial in $s = i\omega$ for the reduced frequency k at the flutter point

$$(C' + iC'') = a_0 + a_1 s + a_2 s^2 \tag{5}$$

and introducing the actuator transfer function as

$$\frac{\Delta\beta}{x_i} = F_{ACT} = \frac{1}{1 + b_1 s + b_2 s^2} \tag{6}$$

which provides the necessary condition for the added control degree of freedom $\Delta\beta$ then the state-space-description of (1) is as follows:

$$\{\dot{x}\} = [A]\{x\} + \{B\} x_i \tag{7}$$

where

$$\{x\} = \begin{Bmatrix} q \\ \beta_0 \\ \Delta\beta \\ \dot{q} \\ \dot{\beta}_0 \\ \Delta\dot{\beta} \end{Bmatrix} = \text{STATE VECTOR} \tag{8}$$

$$[A] = \begin{vmatrix} 0 & I \\ [S_1]^{-1} \bullet [S_2] & [S_1]^{-1} \bullet [S_3] \end{vmatrix} \tag{9}$$

$$[S_1] = \left[-\frac{1}{\omega_r^2} \begin{vmatrix} M_{qq} & M_{q\beta_0} & M_{q\Delta\beta} \\ M_{\beta_0 q} & M_{\beta_0\beta_0} & M_{\beta_0\Delta\beta} \\ 0 & 0 & \omega_r^2 b_2 \end{vmatrix} + \frac{\rho}{2} v^2 \frac{F}{s_R} \frac{1}{m_r \omega_r^2} \begin{vmatrix} a_{2qq} & a_{2q\beta_0} & a_{2q\Delta\beta} \\ a_{2\beta_0 q} & a_{2\beta_0\beta_0} & a_{2\beta_0\Delta\beta} \\ 0 & 0 & 0 \end{vmatrix} \right] \tag{10}$$

$$[S_2] = \left[\begin{vmatrix} K_{qq} & 0 & 0 \\ 0 & K'_{\beta_0\beta_0} & 0 \\ 0 & 0 & 1 \end{vmatrix} + \frac{\rho}{2} v^2 \frac{F}{s_R} \frac{1}{m_r \omega_r^2} \begin{vmatrix} a_{0qq} & a_{0q\beta_0} & a_{0q\Delta\beta} \\ a_{0\beta_0 q} & a_{0\beta_0\beta_0} & a_{0\beta_0\Delta\beta} \\ 0 & 0 & 0 \end{vmatrix} \right] \tag{11}$$

$$[S_3] = \left[\frac{s_R}{v k} \left(\begin{vmatrix} gK_{qq} & 0 & 0 \\ 0 & K''_{\beta_0\beta_0} & 0 \\ 0 & 0 & \frac{v_k}{s_R} b_1 \end{vmatrix} + \frac{\rho}{2} v^2 \frac{F}{s_R} \frac{1}{m_r \omega_r^2} \begin{vmatrix} a_{1qq} & a_{1q\beta_0} & a_{1q\Delta\beta} \\ a_{1\beta_0 q} & a_{1\beta_0\beta_0} & a_{1\beta_0\Delta\beta} \\ 0 & 0 & 0 \end{vmatrix} \right) \right] \tag{12}$$

and

$$\{B\} = \begin{Bmatrix} 0 \\ 0 \\ 0 \\ [S_1]^{-1} \cdot \begin{Bmatrix} 0 \\ 0 \\ 1 \end{Bmatrix} \end{Bmatrix} \tag{13}$$

To get the optimal control law K_{opt} the quadratic performance criterion is minimized

$$J = \int_0^\infty \left(\{x\}^T [Q] \{x\} + x_i R x_i \right) dt \tag{14}$$

Q is a weighting matrix found by trial and error on using a screen together with the computer. R is a scalar, to be selected, because there is only one control surface.

Minimizing (14) leads to the optimal control laws

$$x_i = \{K_{opt}\}^T \{x\} \tag{15}$$

with

$$\{K_{opt}\}^T = - R^{-1} \{B\}^T [P] \tag{16}$$

where P is the steady state solution of the Matrix Riccati equation

$$[-P] = [P][A] + [A]^T[P] - [P]\{B\} R^{-1} \{B\}^T [P] + [Q] \tag{17}$$

In our case the complete state vector can be measured and fed back:

The flutter system can be described by two modes q,

I first wing bending

II first wing torsion + store pitch

These modes can be measured by two accelerometers on the wing the signals of which will be added and subtracted.

The aileron motion can be described by

$$\frac{m_\beta}{K_\beta} = \beta_0 \tag{18}$$

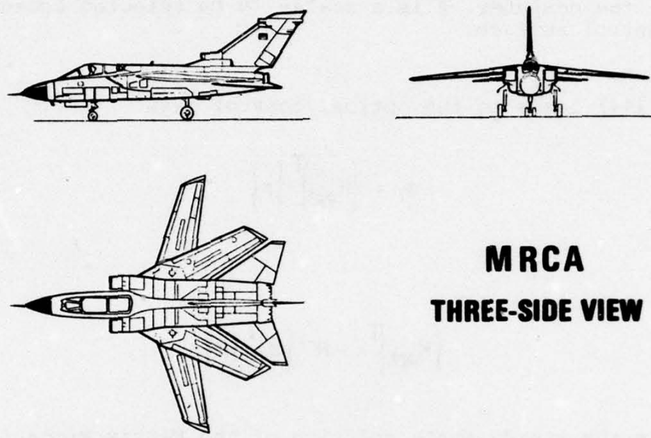
where m_β = hinge moment measured by strain gauge

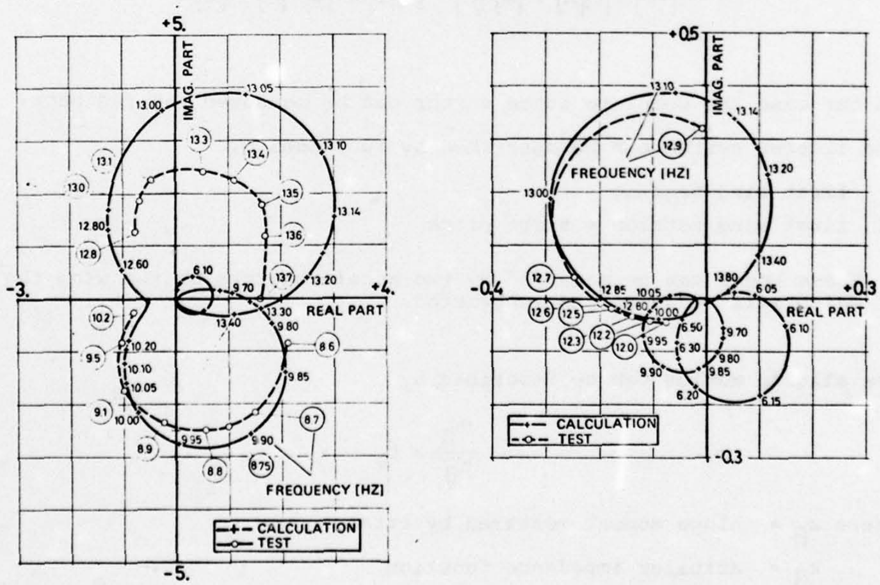
k_β = actuator impedance function

2. IMPACT OF ACTIVE CONTROL ON STRUCTURES

This chapter deals with the analysis methods and experimental procedures which have to be used when implementing ACT into an elastic aircraft (and all airplanes are elastic!). These methods can be used to layout the system and to assure against detrimental coupling of the structure with the control system.

Three newly designed fighter or combat airplanes have been reported to have exhibited some kind of the above mentioned interaction [26], [27], [28]. We shall restrict ourselves just to describe the investigation procedures used on the Tornado (being similar to the ones used in the other references) and want to point out, that the normally used concept of implementing notch filters into the control system is no more applicable when ACT-functions requiring the higher frequency regime such as vibration mode control, flutter mode control are installed.





UNSTABLE SYSTEM

STABLE SYSTEM with NOTCH FILTER

FIG. 31 NIQUIST DIAGRAM TO CHECK CSAS – STRUCTURAL MODE COUPLING

Several input data to form mathematical model must be determined or predictions must be checked and if necessary corrected by tests. These tests are: Ground Resonance Test to check the structure elastic behaviour, transfer and impedance functions for the hydraulic actuators for several parameters (because of nonlinearities).

Open loop tests on the complete system - aircraft-control-system - must be done and stability must be assured before the electrical loops can be closed. Niquist diagrams for the Tornado - being the first operational aircraft that features a triplex analogue fly-by-wire system and automatic stabilization - are shown in Fig. 31. This figure shows an unstable structural mode because the point -1 is encircled clockwise. After a notch filter was implemented the system was stable. If other ACT-functions than damping rigid body modes would have been applied (requiring the higher frequency regime) then the stabilization of this structural mode would have required other means like relocation of rate gyros.

Flight tests with the engaged control system have to be done as well to assure stability throughout the whole flight regime. All these investigations will become more complex with increasing number of interconnected ACT-functions.

CONCLUSIONS

The usage of active control technology offers considerable gains for structural design. Some technologies must be implemented directly into the design to give the optimum benefit. Other technologies can be retrofitted assuming that the aircraft was designed with a sufficient number of independently controllable surfaces.

The aeroelastic analysis and test methods must be extended to contain the active control functions and their impact on structures. Optimal control theory is advocated for the active suppression of complicated flutter modes.

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STRUCTURES NOUVELLES EN MATERIAUX COMPOSITES A HAUTES PERFORMANCES POUR AVIONS DE COMBAT.

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1.- RESUME -

Cet exposé comprend trois parties :

- Dans la première partie, après un rappel des objectifs visés et une analyse succincte de la politique de l'AEROSPATIALE dans le domaine des nouvelles structures en matériaux composites, sont décrites les principales techniques utilisées pour l'élaboration des structures nouvelles en matériaux composites.
- Dans la deuxième partie, est présenté l'ensemble des principales réalisations de l'AEROSPATIALE dans le domaine des structures en matériaux composites à hautes performances.
- Dans la troisième partie, en fonction de l'expérience acquise dans le domaine des composites, sont analysés les principes constructifs qui pourraient être adoptés pour la réalisation de la structure d'un avion de combat.

Dans cette présentation, l'ensemble de la structure de l'avion est découpé en un certain nombre d'éléments principaux pour lesquels sont définis, après discussion des avantages et des inconvénients, les types de structure les plus appropriés en mentionnant la nature des matériaux et les procédés de fabrication utilisés pour leur réalisation.

En conclusion, nous présenterons, en comparaison qualitative avec l'avion de base en structure classique alliage léger, les différentes possibilités de versions d'avion de combat permises par la réduction de masse liée à l'utilisation des matériaux composites.

2.- INTRODUCTION -

On peut dire aujourd'hui qu'après une phase de recherche et de mise au point en laboratoire, suivie d'une phase de réalisation d'éléments de structure prototypes, les matériaux composites à matrices organiques renforcées de fibres à hautes performances ont atteint le début de l'industrialisation.

Les caractéristiques élevées de ces nouveaux matériaux et les procédés de fabrication mis au point pur leur élaboration permettent de penser que, dans un avenir proche, leur emploi se développera rapidement dans le domaine des structures d'avionscivils et militaires.

2.1 - Objectifs -

Le remplacement des structures classiques en alliage léger, par ces nouvelles structures en matériaux composites permet d'obtenir un gain de masse de l'ordre de 15 à 30 %, entraînant une économie de carburant et une amélioration sensible des performances. Jusqu'à présent, le coût de réalisation de ces nouvelles structures s'avérait souvent trop élevé par rapport aux taux d'échange (masse-prix) fixés pour chaque type d'appareil, en particulier pour les avions civils pour lesquels cette notion est relativement plus accessible que pour les appareils militaires.

La décroissance du prix d'achat des matériaux et la mise au point de nouvelles techniques de fabrication, comme l'emploi des tissus préimprégnés et des outillages en silicone thermoexpansible permettent aujourd'hui de penser qu'il est possible de réaliser des structures en matériaux composites à un coût proche de celui des structures alliage léger et parfois même à un coût inférieur. En outre, ces nouvelles structures entraînent un gain sur les coûts de maintenance par diminution ou absence de problèmes de corrosion et de fatigue.

2.2 - Matériaux utilisés -

En ce qui concerne les différents matériaux présentés sur le marché, après un certain nombre d'essais comparatifs, le choix de l'AEROSPATIALE s'est porté en faveur de la fibre de carbone, dont les propriétés fournissent le meilleur compromis technico-économique. L'emploi du bore étant réservé aux renforcements locaux de certaines structures.

Le Kevlar est, pour le moment, réservé à des éléments peu travaillants ou à des structures sollicitées exclusivement en traction, cas assez peu fréquent dans une cellule d'avion.

Les premiers éléments fabriqués ont été des structures simples, dites "secondaires" : (panneaux de Karman, carénages, portes de visite). Après une mise au point des procédés de fabrication, de nouvelles structures plus importantes (type gouvernes) ont été étudiées et fabriquées, pour aboutir, aujourd'hui, à l'étude et à la réalisation de structures primaires comme les empennages ou les caissons de voilure (en projet).

La plupart des éléments étudiés et fabriqués sont soumis à des essais statiques jusqu'à rupture, à des essais de fatigue et de vieillissement et souvent à des essais d'expérimentation en vol.

2.3 - Types de structures composites utilisés -

Le premier type de structure étudié par l'AEROSPATIALE est le sandwich collé (fig. 1) ; il comporte deux revêtements en composite collés sur une âme en nid d'abeilles et un encadrement métallique ou composite.

Deux techniques sont utilisées pour cette fabrication :

- pour les pièces importantes supportant des charges élevées, les éléments en carbone (revêtements, longerons, nervures) sont polymérisés séparément et collés ensuite entre eux et avec l'âme nid d'abeilles.
- pour les pièces secondaires, leur réalisation s'effectue en une seule opération de moulage (procédé co-curing). Les performances de ces structures sont alors inférieures aux précédentes mais d'un coût de fabrication plus faible.

L'introduction des efforts importants est assurée par des ferrures en titane intégrées dans la structure composite par collage. En effet, la différence trop grande entre les coefficients de dilatation de l'alliage léger et du carbone empêche l'utilisation de ferrures en alliage léger pour ce type d'assemblage.

De nombreux essais ont été réalisés pour mettre au point le collage titane-carbone.

Pour des raisons de facilité de collage des revêtements, d'insensibilité totale à la corrosion et de meilleure tenue à la foudre, les nids d'abeilles nomex ou ASK ont été préférés aux nids d'abeilles métalliques pour constituer l'âme des structures sandwich, malgré leur moins bonnes performances dans leur rôle de stabilisateur des revêtements comprimés. Le nid d'abeilles nomex est un film en polyamide aromatique imprégné de résine formophénolique et le film ASK est un film à base de fibres de cellulose imprégné de résine formophénolique récemment développé par l'AEROSPATIALE pour les structures économiques.

Le deuxième type de structure est dit "monolithique". Cette structure se rapproche davantage de la structure classique.

Elle peut être assemblée par collage, par fixations mécaniques, ou encore obtenue par moulage "intégral" (fig. 2.)

Alors que pour les structures collées l'utilisation de ferrures en titane s'impose, pour celles assemblées par fixations mécaniques, l'emploi de ferrures en alliage léger est rendu possible en prenant, malgré tout, quelques précautions contre la corrosion dûe au couple galvanique entre le carbone et l'alliage léger. Cela permet d'abaisser les coûts de production et, ainsi, de rendre plus compétitive la solution composite.

Deux problèmes importants sont apparus dès les premières réalisations d'éléments en structures composites :

- l'introduction d'efforts concentrés,
- la tenue à la foudre.

2.4 - Principes de renforcements locaux -

Pour l'introduction des efforts concentrés, plusieurs solutions ont été étudiées et mises au point (Fig. 3).

Le premier procédé consiste à introduire l'effort par l'intermédiaire d'une pièce en titane collée par "steps" sur la structure composite. Ce procédé est coûteux, mais assure une mise en charge correcte et progressive des revêtements.

Le deuxième procédé consiste à réaliser un renforcement local des revêtements pour diminuer les contraintes de matage au droit des fixations.

Ce renforcement peut être obtenu par un épaississement de la pièce au droit de l'introduction de l'effort par adjonction de plis supplémentaires de carbone ou par interposition de plis de bore ou de lamelles de titane dans les revêtements en carbone.

2.5 - Protection contre la foudre -

La tenue à la foudre est un problème très important, en particulier pour les caissons réservoirs.

En effet, si les parois de carbone sont suffisamment conductrices pour écouler l'électricité statique ou permettre le fonctionnement des antennes, elles supportent moins bien que l'alliage léger le passage des forts courants de foudre.

Pour des éléments vitaux de l'avion exposés à la foudre, certaines précautions doivent donc être prises pour améliorer la métallisation. Un encadrement métallique parfaitement relié au reste de la structure peut parfois être suffisant.

Par contre, pour les caissons réservoirs, un grillage métallique étalé et collé en surface sur les revêtements au cours de la polymérisation paraît nécessaire pour assurer l'écoulement parfait des charges électriques de la foudre avec, en outre, une peinture conductrice en surface.

Tous les becs d'attaque, zone privilégiée pour les impacts de la foudre, doivent comporter à leur extrèmité avant un élément métallique.

3.- PRESENTATION DES PRINCIPALES ETUDES ET REALISATIONS DE LA DIVISION AVION DE L'AEROSPATIALE -

3.1 - Structures dites "secondaires" -

Toutes ces structures sont du type "sandwich". Elles sont constituées de 2 revêtements carbone collés sur une âme en nid d'abeilles Nomex ou ASK. Les bordures sont protégées par un renforcement carbone ou titane. Les revêtements sont polymérisés séparément et collés ensuite sur le nid d'abeilles ou assemblés par polymérisation "in situ".

Les ferrures d'accrochage sont fixées sur l'élément par l'intermédiaire de vis prenant réaction dans des inserts intégrés dans la structure par collage.

Principales réalisations (voir photos) -

- Porte de soute radio de CORVETTE,
- Porte de nacelleVAUTOUR,
- Carénages de servo-commande CONCORDE,
- Trappes de train principal CONCORDE,
- Structure Karman de voilure CONCORDE,
- Porte de visite Karman de voilure CONCORDE.

3.2 - Gouvernes -

Comme les structures secondaires, les premières gouvernes réalisées par l'AEROSPATIALE sont du type sandwich. Les ferrures principales d'introduction d'efforts en titane sont intégrées dans l'ensemble de la structure et collées aux revêtements et au nid d'abeilles. Les ferrures secondaires peuvent être en alliage léger et rapportées par vis.

Actuellement, de nouvelles techniques de fabrication utilisant les propriétés thermoexpansibles du silicone ont été mises au point et permettent, avec des coûts plus faibles que pour le sandwich, de réaliser des gouvernes en structure monolithique.

Principales réalisations (voir Photos) -

- Elevons internes CONCORDE,
- Elevons externes CONCORDE,
- Lift dumper AIRBUS.

3.3 - Structures primaires -

Pour ces éléments, les deux types de structures sandwich et monolithique sont utilisés. Mais jusqu'à présent, à l'AEROSPATIALE, en fonction des éléments étudiés, les premières ont été les plus employées. Actuellement, la réalisation d'un caisson de voilure, type réservoir de combustible, impose une structure monolithique, constituée de revêtements raidis fixés sur une ossature "creuse".

Le procédé de moulage par silicone thermoexpansible permet la fabrication de telles structures à des coûts compétitifs avec la version alliage léger.

Ce type de structure sera adopté également pour abaisser les coûts de fabrication de certains éléments réalisés jusqu'alors en sandwich.

Principales réalisations - (voir Photos)

- Saumon de voilure CONCORDE Développement,
- Dérive MIRAGE 2000 pour les Avions Marcel DASSAULT,
- Panneau de revêtement voilure CONCORDE,
- Voilure engin-cible,
- Bielles.

4.- STRUCTURE D'UN AVION DE COMBAT EN MATERIAUX COMPOSITES -

Par ses dimensions, sa définition même, par l'importance de ses facteurs de charges conduisant à des dimensionnements élevés, l'avion de combat est l'appareil pour lequel l'utilisation des nouvelles structures composites s'avère la plus avantageuse. Il est, d'autre part, plus facile d'expérimenter des pièces en vol sur un avion de combat que sur un appareil civil.

Les connaissances acquises aujourd'hui nous permettent d'envisager la fabrication d'un avion de combat avec une structure réalisée à 60 % environ en matériaux composites à hautes performances.

Les principaux éléments prévus en composites sur le projet d'avion de combat étudié par l'AEROSPATIALE (figures 4 & 5) sont représentés sur la figure 6 :

- le caisson principal de voilure,
- l'onglet de voilure,
- la dérive,
- le fuselage avant,
- les élevons,
- les trappes de train,
- le volet d'onglet,
- les aérofreins,
- les carénages divers.

Dans ce paragraphe, nous présenterons les principes constructifs retenus pour la réalisation des principaux éléments en matériaux composites pouvant être utilisés pour un avion de combat.

4.1 - Voilure -

4.1.1 - Caisson principal -

Cet élément essentiel de l'avion, fortement chargé et généralement utilisé pour le stockage du combustible se conçoit obligatoirement comme un caisson constitué par deux revêtements raidis assemblés sur une ossature longerons-nervures.

L'option principale, qui déterminera l'architecture interne du caisson, est le mode de liaison au fuselage.

En construction classique, on trouve ici l'alternative :

- Fixation ponctuelle par deux, trois points ou même davantage, correspondant chacun à un longeron du caisson et à un cadre du fuselage,
- Fixation continue par un éclissage. Le revêtement du caisson trouve alors un répondant de même profil, dans la traversée du fuselage.

Chacune de ces deux versions trouve, dans l'aviation mondiale, ses partisans, de telle sorte qu'aucune ne s'est imposée de toute évidence, les méthodes modernes de fabrications métalliques permettant de traîter aussi rationnellement l'une et l'autre.

Dans l'état présent de nos connaissances, la construction composite ne permet pas, en toute sécurité et en toute évidence, de considérer au départ les deux solutions sur le même plan de compétition.

En effet, la fixation continue exige un éclissage, généralement assuré, pour des avions de combat, par des boulons en traction (fig. 7) que la technologie actuelle ne permet pas d'intégrer au revêtement en composite.

Cette liaison exige donc une éclisse métallique intermédiaire que l'on sait réaliser, par exemple, en titane, avec un assemblage au composite collé par steps, au prix, toutefois, d'une fabrication d'autant plus coûteuse et délicate que le profil d'emplanture sera de plus grande profondeur.

La solution par attaches ponctuelles, qui ramène les contraintes des revêtements sur les longerons par des liaisons travaillant en cisaillement, directement pratiquées dans le composite apparaît aujourd'hui infiniment plus sûre et moins coûteuse.

C'est donc cette solution que nous avons adoptée dans ce projet d'avion de combat.

Le caisson est fixé en quatre points. Les attaches avant et arrière ne contribuent qu'au passage des efforts tranchants verticaux (fig. 8).

L'ossature interne sera donc du type à quatre longerons principaux, et, à nouveau, deux principes constructifs peuvent être envisagés :

Premièrement : Le caisson est constitué de 2 revêtements d'épaisseurs évolutives fixées sur une ossature composée de nombreux longerons et peu de nervures.

Deuxièmement : Le caisson est constitué de 2 revêtements raidis par lisses rapportées ou intégrées, l'ossature ne comportant que quelques longerons et un nombre de nervures relativement important.

Ce dilemme qui se pose avec les structures métalliques ne peut être tranché par l'emploi des matériaux composites.

Seule, peut-être, une analyse fine des masses et des coûts peut permettre de faire un choix.

Remarque - En ce qui concerne la définition des revêtements, nous avons adopté la solution monolithique, de préférence à celle des panneaux sandwich pour plusieurs raisons :

1 - L'épaisseur de ces derniers entraîne une diminution de la capacité des réservoirs de combustible,

2 - Du fait de la présence de 2 revêtements dans un panneau sandwich, celui de l'extérieur est forcément plus mince que s'il s'agissait d'une structure monolithique. Cela est défavorable pour la protection contre les impacts et aussi contre la foudre.

3 - Si, en section courante, les panneaux sandwich présentent une incontestable simplicité, il n'en est pas de même au droit des éclissages et des introductions d'efforts.

4 - Coût élevé de ces structures du fait du nombre d'éléments qui les constituent et du nombre d'opérations nécessaires à leur fabrication.

5 - Enfin les structures sandwich collées présentent de plus grandes difficultés d'étanchéité.

Nous retiendrons la structure sandwich pour des éléments bien particuliers et principalement pour des gouvernes de profil mince constituées, pour des raisons de rigidité ou de résistance aux endommagements, d'une âme traversante en nid d'abeilles.

L'utilisation des matériaux composites avec les procédés de mise en oeuvre relativement simples qu'ils permettent et notamment l'emploi tout récent des outillages en silicone thermoexpansible, laisse envisager la fabrication de caissons entiers en une seule opération de moulage et de polymérisation.

Quand on sait la part prépondérante prise par la multiplicité des pièces et des assemblages dans l'évaluation du coût de production, il est certain que la réalisation d'une telle structure permettrait d'obtenir des coûts de production remarquablement bas, malgré les prix plus élevés des matériaux.

Cependant, en fonction de nos connaissances actuelles, et pour des raisons de maintenance, d'inspection, de contrôle en fabrication, de facilité de mise en place de certaines ferrures et de remplacement éventuel d'éléments endommagés, nous présentons le caisson de voilure constitué d'un ensemble de pièces liées entre elles par fixations mécaniques.

Les possibilités des élastomères ne sont donc pas utilisées au maximum comme dans la fabrication du caisson en une seule opération. Mais l'ensemble des éléments constituant la structure du caisson est réalisé par moulage avec les outillages en silicone thermoexpansible. Cela diminue, malgré tout, le nombre de pièces par rapport à la structure classique alliage léger et entraîne une baisse sensible des coûts de production qui s'ajoute à celle obtenue par l'emploi le plus généralisé possible des tissus imprégnés à la place des nappes unidirectionnelles.

Le principe constructif adopté permet, en outre, l'utilisation de ferrures ou tout autre élément en alliage léger, alors que la conception monobloc ou collée imposerait l'emploi de ferrures en titane dont le coefficient de dilatation thermique $\delta = 9.10^{-6}$ est plus proche de celui des matériaux composites $\delta = 0$ que celui de l'alliage léger ($\delta = 21.10^{-6}$). Le remplacement du titane par l'alliage léger permet ainsi d'obtenir une diminution supplémentaire des coûts de fabrication.

Comme nous l'avons vu dans la présentation générale des éléments en matériaux composites, il existe deux principes d'introduction des efforts dans ces structures :

- directement par fixations mécaniques,
- indirectement par l'intermédiaire d'une plaque renfort titane intégrée par collage.

Ce principe a été mis au point par l'AEROSPATIALE, et il offre certainement le meilleur rendement mécanique. Cependant, pour des raisons de coûts, nous adopterons les liaisons par fixations mécaniques, placées directement dans la paroi composite renforcée localement.

Tenant compte des descriptions précédentes, la définition du caisson est présentée Fig. 9.

Cet élément est constitué de 2 revêtements en fibres de carbone monolithique moulés en une seule opération avec les raidisseurs intégrés. Les revêtements sont assemblés par fixations mécaniques sur une ossature comprenant un encadrement métallique, sur lequel sont accrochés les becs d'attaque et les élevons, un ensemble longerons et nervures en structure monolithique auto-raidie elle-même en composite carbone réalisée également par moulage au silicone (Fig. 10).

Au droit des liaisons sur les longerons principaux et les ferrures d'introduction d'efforts métalliques, des renforcements locaux des revêtements sont obtenus par adjonction de plis de carbone supplémentaires ou de plis de tissus de bore.

La protection contre la foudre est assurée par :

- l'encadrement métallique, lequel est relié électriquement au fuselage,
- un grillage métallique placé sur la surface des revêtements au moment de leur polymérisation,
- une peinture conductrice.

En outre, les becs d'attaque, zone de grande probabilité d'impact de la foudre sont prévus métalliques.

L'étanchéité du réservoir est obtenue, comme pour les structures classiques en alliage léger, par interposition de mastic au droit des différentes zones d'accostage et sur les têtes des fixations.

4.1.2 - Onglet - (Fig. 11) -

Cet élément de voilure, à l'avant du caisson principal, est caractéristique de la formule aérodynamique adoptée. Il ne supporte pas de charges élevées et est traîté uniquement comme réservoir, accorché en trois points sur le fuselage par l'intermédiaire de ferrures en alliage léger. Il s'agit d'un caisson constitué de deux revêtements en composites fibres de carbone, monolithique à raidisseurs intégrés fixés sur une ossature composée d'un encadrement métallique et de nervures internes en composites fibres de carbone auto-raidies.

4.1.3 - Surfaces mobiles -

Elevons - (Fig. 12) - Pour ces éléments de profils minces, le type de structure sandwich intégral s'impose. Cette structure comprend deux revêtements composites collés sur une âme nid d'abeilles Nomex ou ASK et un encadrement constitué d'éléments en composites ou en titane au droit des fortes introductions d'efforts.

Pour abaisser les coûts de fabrication, les préimprégnés tissus seront le plus souvent utilisés à la place des nappes unidirectionnelles.

Volets d'onglet - (Fig. 13) - En ce qui concerne cet élément situé à l'avant de la voilure, 2 types de structures sont envisageables :

- sandwich intégral, comme pour les élevons,
- monolithique obtenue par moulage intégral : l'utilisation des outillages en silicone thermoexpansible permet de réaliser en une seule opération cet élément constitué de 2 revêtements moulés "in situ" avec l'ensemble des nervures internes de raidissement. Les ferrures d'introduction d'efforts en alliage léger sont rapportées par fixations mécaniques.

Le bec d'attaque sera protégé contre l'érosion et les chocs par un becquet en titane ou inox collé. Pour la même raison, les revêtements comporteront sur leur surface externe un ou deux plis de tissus en fibres de Kevlar, moins sensibles à l'érosion et aux chocs que le carbone.

4.2 - Dérive - (Fig. 14) -

Sur le projet d'avion à commandes de vol électriques présenté par l'AEROSPATIALE, la dérive ne comporte pas de gouvernail.

C'est l'ensemble de la dérive qui est mobile.

Cet élément de profil mince est constitué d'une structure type sandwich comprenant :

- 2 revêtements en fibres de carbone polymérisés et collés sur une âme en nid d'abeilles Nomex ou ASK et sur un encadrement en titane.

Les reprises d'efforts du tourillon sur le revêtement se font par boulonnage direct de la ferrure dans le carbone renforcé localement.

Le bec d'attaque est partiellement démontable pour permettre le cheminement des fils d'équipements. Il est constitué de deux revêtements carbone collés sur une âme nid d'abeilles Nomex et deux longerons en composites. La protection contre l'érosion et les chocs éventuels est assurée par un ou deux plis de Kevlar placés sur la surface externe des revêtements au moment de la polymérisation et par un becquet métallique collé à la partie avant, assurant, en outre, la protection contre la foudre.

4.3 - Trappes de train - (Fig. 15) -

Ces éléments de structure secondaire sont du type sandwich. Les revêtements en préimprégnés tissu carbone seront moulés et collés "in situ" sur une âme en nid d'abeilles ASK. L'encadrement pourra être protégé contre les chocs par un clinquant métallique collé. Les ferrures d'accrochage en alliage léger seront fixées par vis sur la structure des trappes renforcée localement.

4.4 - Aérofreins - (Fig. 16) -

Cet élément est constitué d'une ossature support en alliage léger, comprenant 2 bras d'articulation et 2 nervures sur lesquels est fixé par boulons un panneau galbé en carbone monolithique d'épaisseur constante, renforcé localement.

4.5 - Structure du fuselage avant - (Fig. 17) -

La structure du fuselage d'un avion de combat peut se décomposer en deux parties de types constructifs différents :

- Une partie AR comportant le logement moteur sur laquelle sont accrochés la voilure, la dérive et le train principal.
  Cette zone du fuselage est constituée d'une structure classique en alliage léger, car sa complexité semble peu compatible avec la conception composite, tout au moins dans l'état actuel de nos connaissances.
- Une partie AV, moins complexe, comprenant le poste de pilotage, les logements du train AV et principal, diverses soutes d'équipements, ainsi qu'une partie des réservoirs carburant.

  Cet élément peut être réalisé avantageusement en matériaux composites. Les parois sont des panneaux sandwich reliés entre eux et à l'ossature interne par fixations mécaniques. Les cloisons servant de séparation de soutes ou de plancher sont elles-mêmes en structure sandwich. Tous ces éléments ont un caractère de démontabilité permettant éventuellement leur remplacement.

  Certains profilés de liaison, ainsi que certains cadres forts sont en structure monolithique, moulés en une seule opération dans des outillages en silicone thermoexpansible. Les autres profilés ou ferrures d'introduction de charges sont en alliage léger.

  Comme pour les autres éléments, afin d'obtenir des structures à bas prix de revient, il est fait usage au maximum des préimprégnés tissus de carbone et du nid d'abeilles ASK pour les éléments sandwich obtenus le plus souvent en co-curing.

  Certaines zones, plus exposées aux chocs que d'autres, recevront une protection particulière (plis de Kevlar en surface, par exemple) ou pourront comporter des éléments en alliage léger. En outre, la transparence aux champs électriques des revêtements carbone impose la présence d'une métallisation sur les parois des soutes d'équipements électriques.

5.- INFLUENCE DE L'UTILISATION DES COMPOSITES SUR LES PERFORMANCES DE L'AVION -

Les études effectuées par l'AEROSPATIALE dans le domaine des avions de combat ont abouti à un projet d'avion polyvalent avec une certaine prédominance de la mission interception sur la mission supériorité aérienne.

Nos connaissances actuelles des caractéristiques des matériaux composites et de leur mise en oeuvre nous permettent d'obtenir, sans modification de géométrie, un gain de masse de 360 Kg environ sur une masse totale de structure de l'appareil basique en alliage léger estimée à 2.724 Kg, dont 62 % environ ont été remplacés par une structure composite (Tableau 1). Ce gain de masse obtenu directement sur certains éléments de la structure entraîne une réduction plus importante sur la masse de calcul de l'avion d'au moins 450 Kg par répercussion sur le dimensionnement des éléments de structures non transformés en composites.

En fonction des objectifs imposés, cette possibilité de gain de masse peut permettre de définir plusieurs versions de l'appareil en partant de la version de base en alliage léger :

- soit en recherchant une augmentation des performances pour améliorer la polyvalence,
- soit, à performances égales, en réduisant la taille de l'appareil et en utilisant un moteur moins puissant.

Quatre hypothèses peuvent être envisagées :

- Sur le tableau 2 sont présentées qualitativement, en comparaison avec la version de base, les caractéristiques de quatre nouvelles versions possibles de l'appareil comportant une structure en grande partie réalisée en matériaux composites.

Version 1 - La géométrie et le moteur de l'avion de base (structure métallique) sont conservés. Dans ce cas, le gain de masse obtenu de 450 Kg entraîne une légère amélioration des performances et du rayon d'action.

Version 2 - Tout en conservant le moteur de l'avion de base et en maintenant approximativement la masse totale de l'appareil au décollage et la masse à vide égales à celles de l'avion de base, la géométrie de la voilure est modifiée : (augmentation de la surface et diminution de l'épaisseur relative).

Le résultat obtenu est une amélioration sensible des performances avec une réduction légère de la consommation en carburant.

Version 3 - Dans cette solution, une augmentation de la surface de voilure est associée à une augmentation de la puissance du moteur. Les gains de performances obtenus seront encors plus importants que dans la version 2, en conservant, malgré tout, une consommation en carburant sensiblement égale à celle de l'avion de base.

Version 4 - Dans cette 4ème version, l'objectif visé est la réduction sensible de la taille de l'appareil à performances égales avec l'avion de base en utilisant un moteur de puissance plus faible. La réduction de la surface de voilure et de la puissance du moteur s'accompagne d'une diminution sensible de la masse de structure de l'avion avec, en outre, une économie de carburant.

6.- CONCLUSION -

Comme nous venons de le voir dans cette présentation, qui synthétise, dans un projet d'avion de combat, une expérience déjà importante de l'adaptation des matériaux composites à l'aviation supersonique, cette mutation de la structure apparaît, aujourd'hui, inéluctable.

En effet, les gains de masse obtenus avec l'utilisation des composites pour un coût de fabrication qui, de plus en plus, devient compétitif grâce à l'emploi de nouvelles techniques de production, vont permettre d'améliorer sensiblement les performances.

En outre, l'expérience acquise sur les avions de combat aura, à son tour, pour conséquence d'étendre l'emploi des matériaux composites dans le domaine des structures d'avions civils, pour lesquels les exigences de calculs et les garanties imposées par les règlements sont plus difficiles à respecter.

BILAN DE MASSE STRUCTURE

TABLEAU 1

ELEMENTS ETUDIES	MASSE (kg) en ALLIAGE LEGER	MASSE (kg) en MATERIAUX COMPOSITES	GAIN de MASSE (kg)
.CAISSON VOILURE	701	537,4	163,6
.ONGLET VOILURE	160	122,4	37,6
.VOLET MOBILE D'ONGLET	25	20,8	4,2
.ELEVONS	79	63,1	15,9
.DERIVE	159	124	35
.TRAPPES DE TRAIN	15	12,8	2,2
.AEROFREINS	15	12,5	2,5
.FUSELAGE AV. : (sans verriere) (sans radome)	549	450	99
TOTAL	1703	1343	360

_MASSE TOTALE STRUCTURE AS X10
AVION DE BASE 2 724 kg

_MASSE DES ELEMENTS ETUDIES EN COMPOSITES

1 703 kg SOIT 62,5% DE LA MASSE TOTALE STRUCTURE

_GAIN DE MASSE REALISE SUR CES ELEMENTS

360 kg SOIT 13% DE LA MASSE TOTALE STRUCTURE OU 21 % DE LA MASSE DES ELEMENTS REALISES en CARBONE

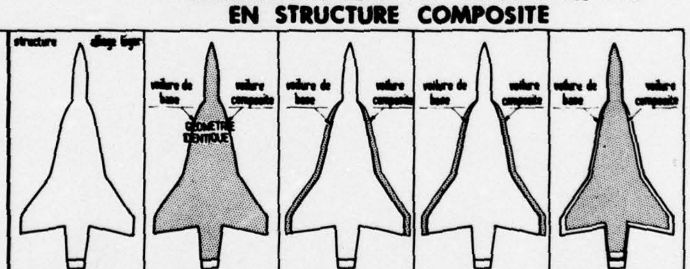
_GAIN DE MASSE AVION

450 kg SOIT 16% TOTAL STRUCTURE

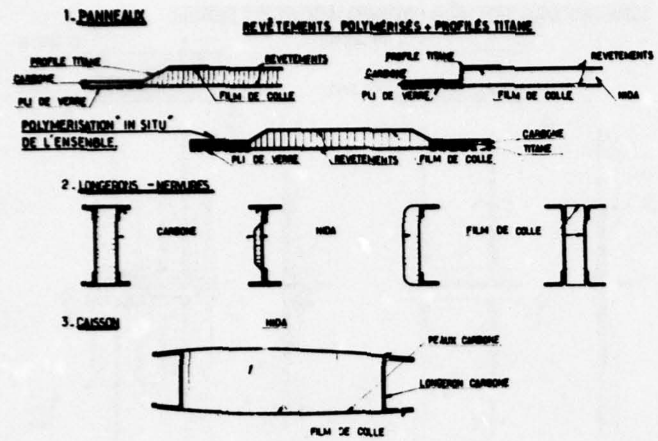
VERSIONS POSSIBLES DE L'AVION AS X10 EN STRUCTURE COMPOSITE

TABLEAU 2

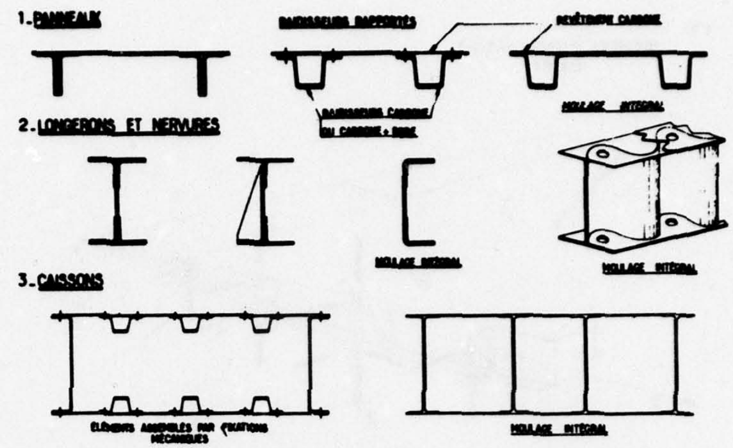
DIFFERENTES VERSIONS



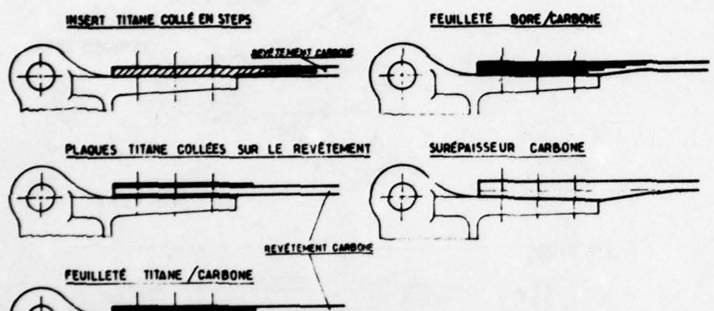
	Version de base	Version 1	Version 2	Version 3	Version 4
PUISSANCE MOTEUR	Q	Q1 = Q	Q2=Q	Q3>Q	Q4<Q
SURFACE VOILURE	S	S1 = S	S2>S	S3>S	S4<S
MASSE AU DECOLLAGE	M	M1=M_450kg	M2 =M	M3 =M	M4<M
FRACTION MASSE STRUCTURE REALISEE EN COMPOSITE	O	0,57	0,62	0,60	0,60
PERFORMANCES EN COMBAT	P	P1>P	P2>P1>P	P3>P2>P	P4=P
CONSOMMATION CARBURANT	C	C1<C	C2<C	C3=C	C4<C



STRUCTURES "SANDWICH" fig. 1



STRUCTURES MONOLITHIQUES fig. 2



PRINCIPES DE RENFORCEMENTS LOCAUX DES STRUCTURES COMPOSITES POUR INTRODUCTION DES EFFORTS

Fig. 3

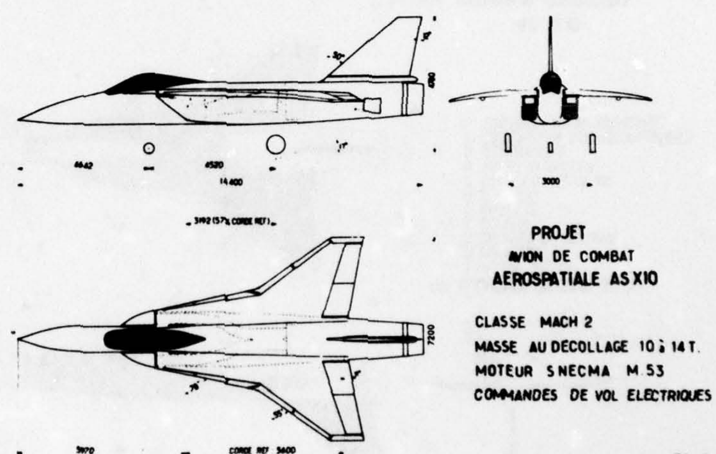
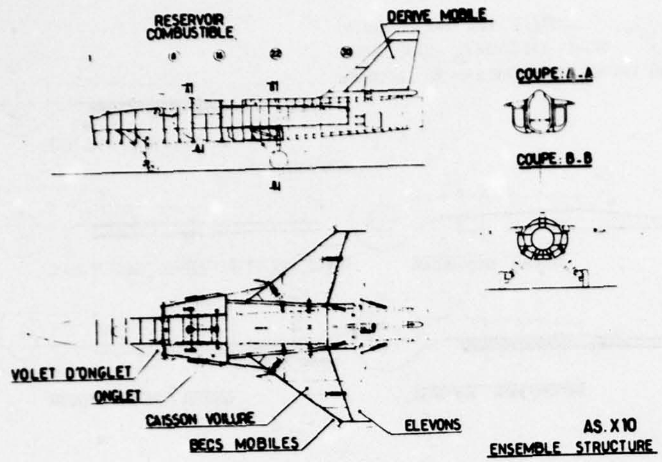
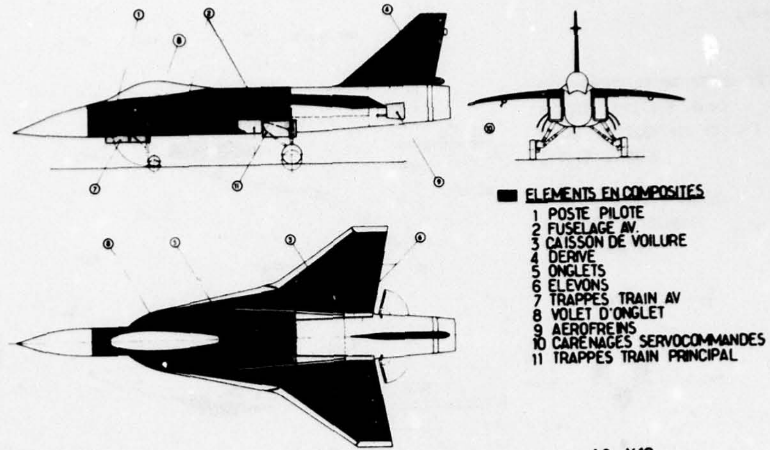


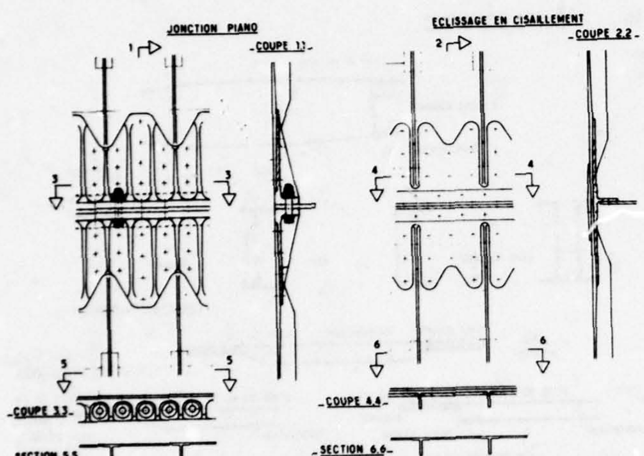
Fig. 4



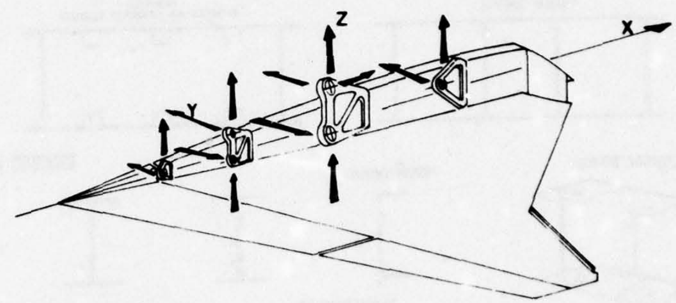
AS. X10
ENSEMBLE STRUCTURE
Fig.5



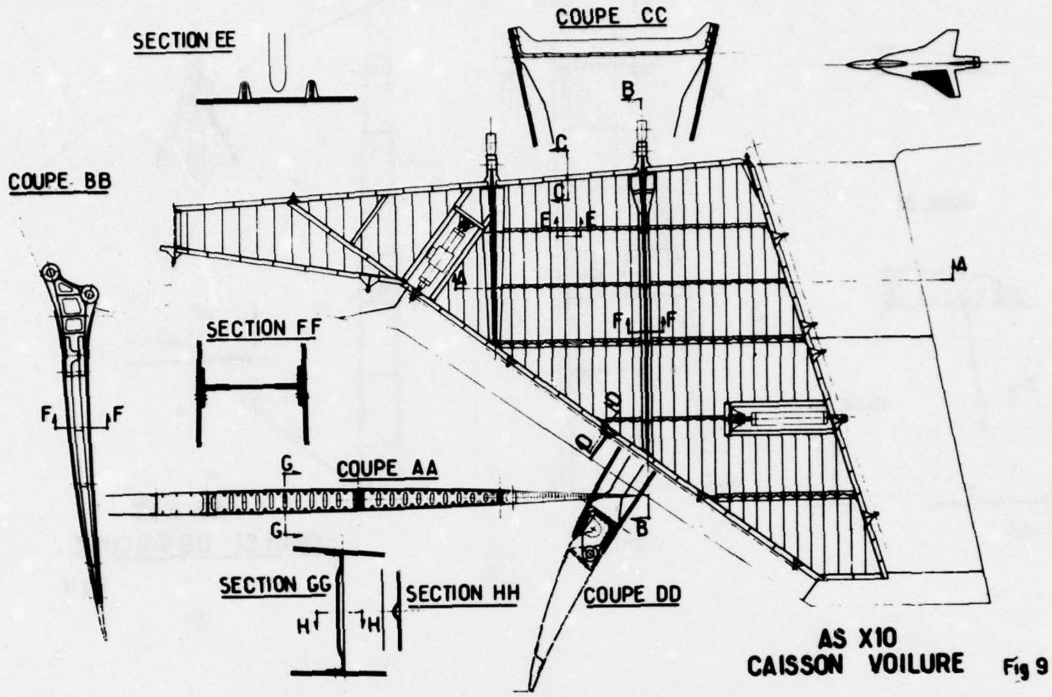
AS. X10
ELEMENTS MATERIAUX COMPOSITES
Fig 6



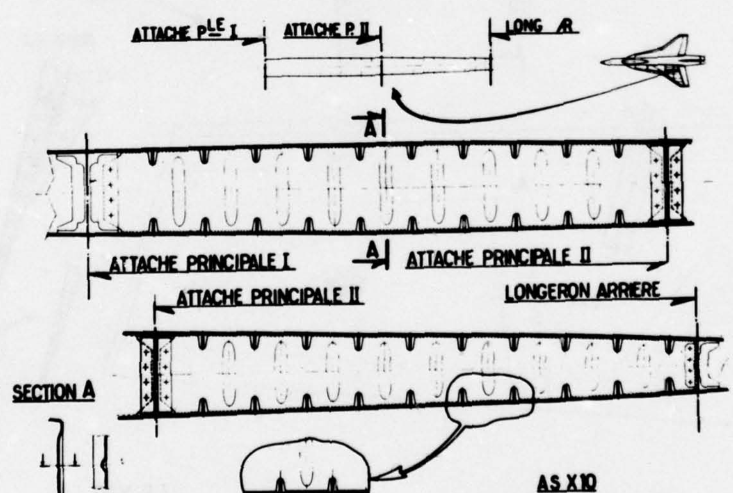
PRINCIPES DE LIAISON (INTRADOS) POUR JONCTIONS CONTINUES Fig 7



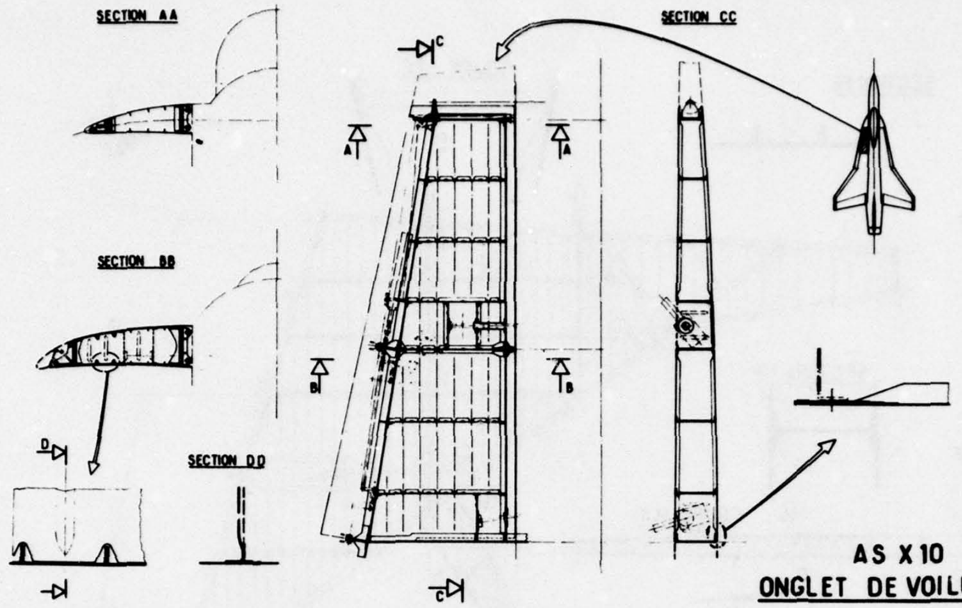
AS. X10
LIAISON VOILURE FUSELAGE
Fig 8



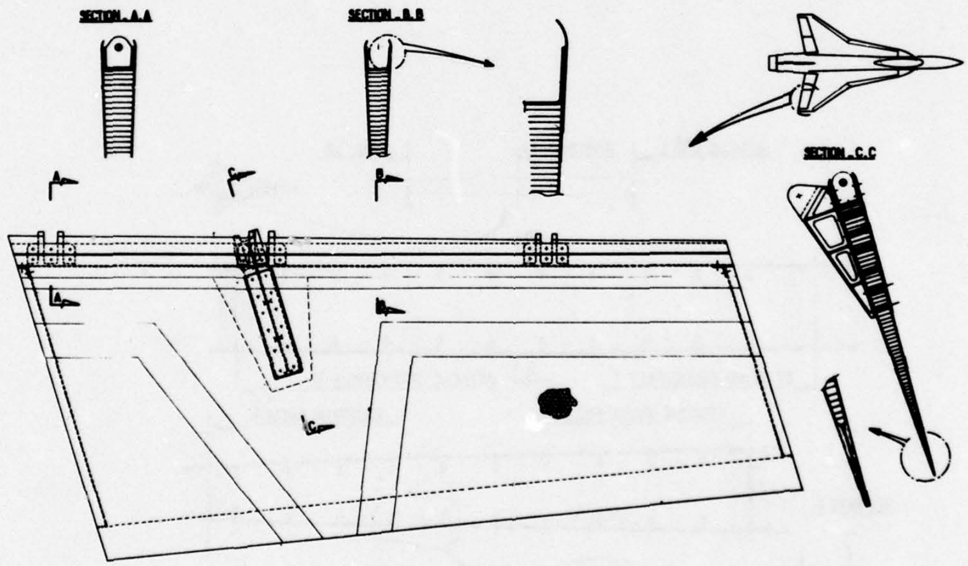
AS X10
CAISSON VOILURE Fig 9



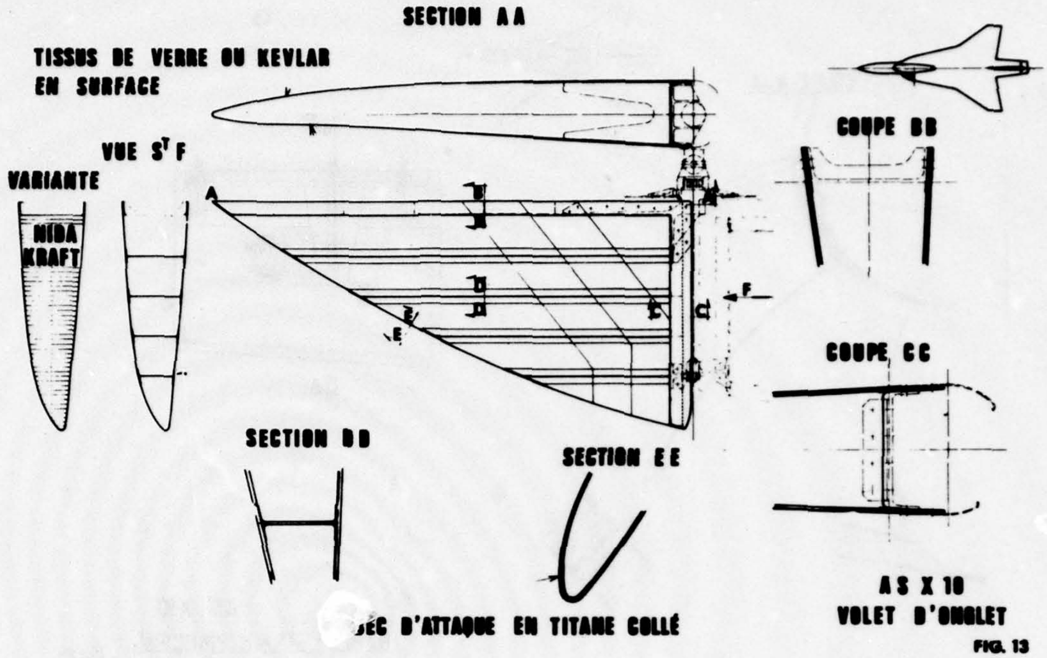
AS X10
NERVURE CAISSON VOILURE Fig.10



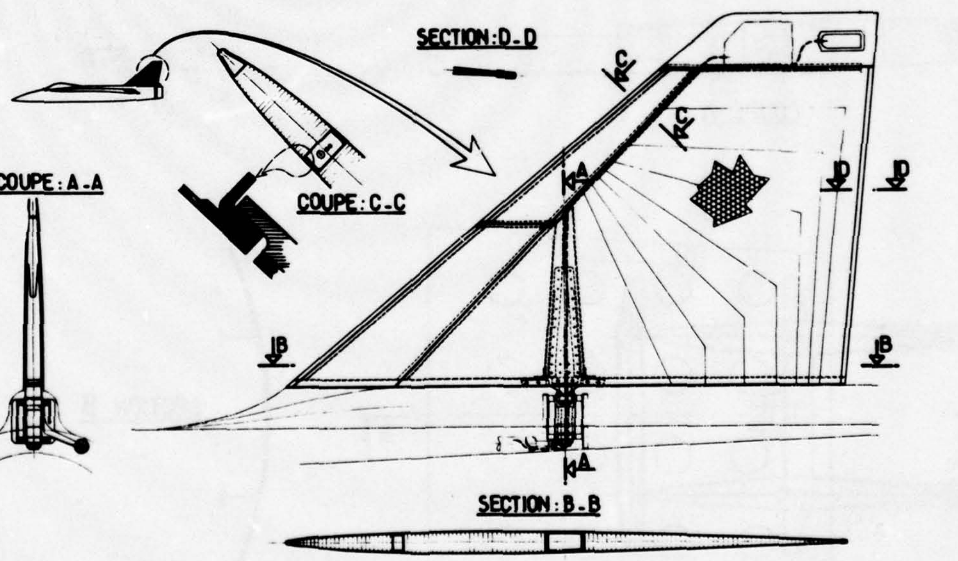
AS X10
ONGLET DE VOILURE
Fig 11



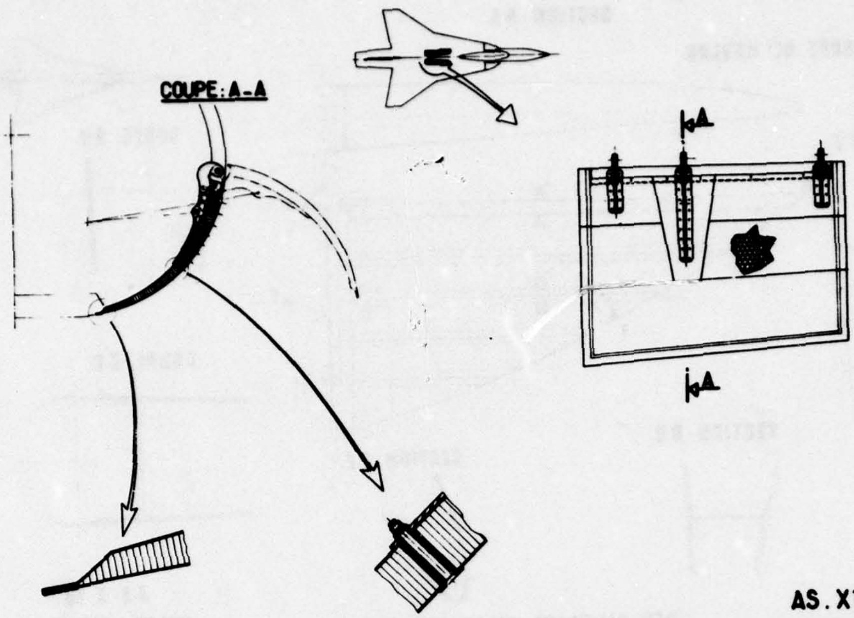
AS X10
ENSEMBLE ÉLEVON
Fig.12



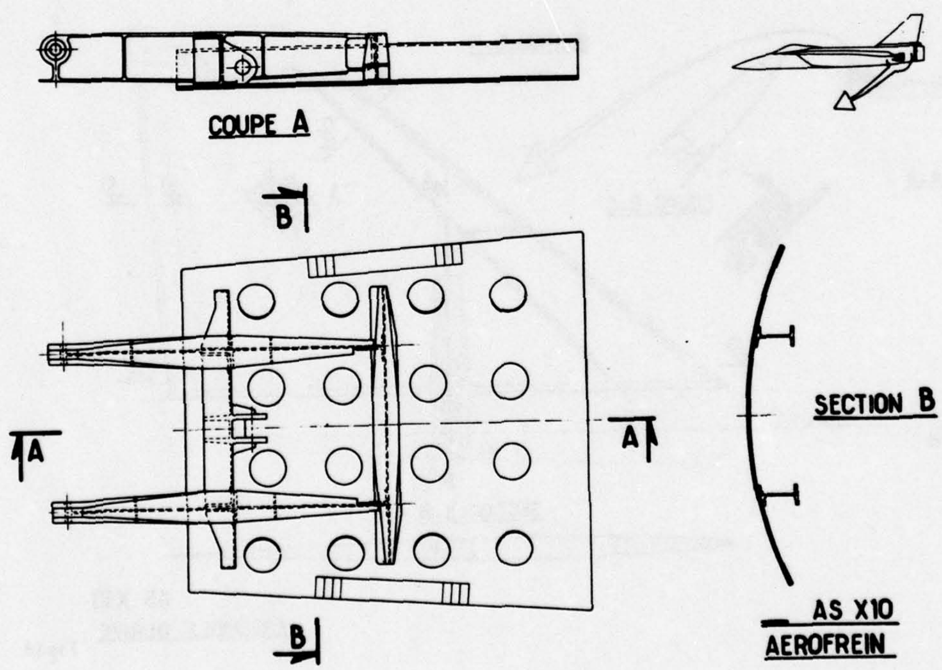
AS X 10
VOLET D'ONGLET
FIG. 13



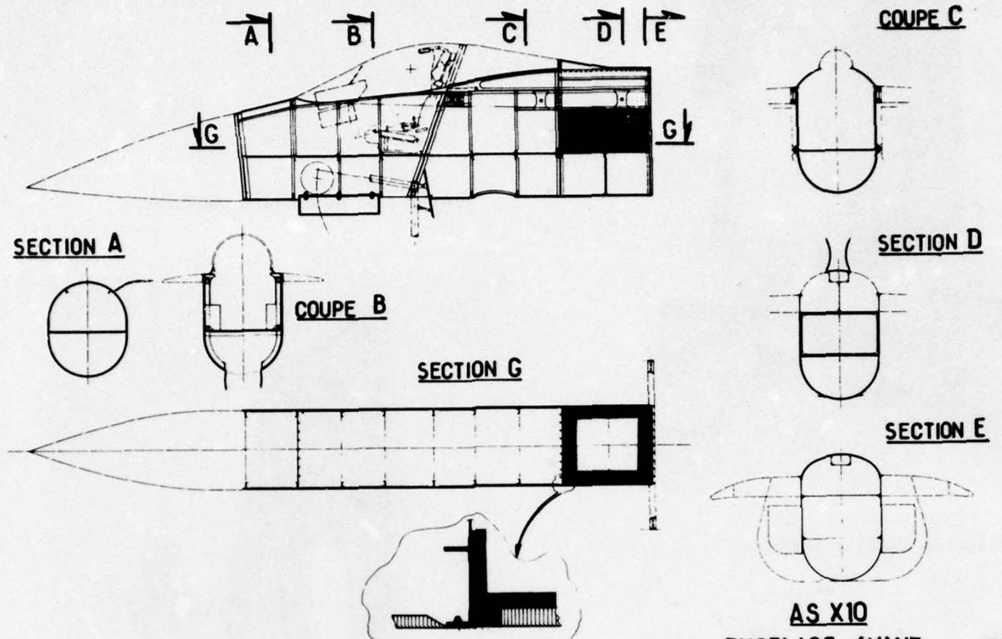
AS.X10
ENSEMBLE DERIVE
Fig 14



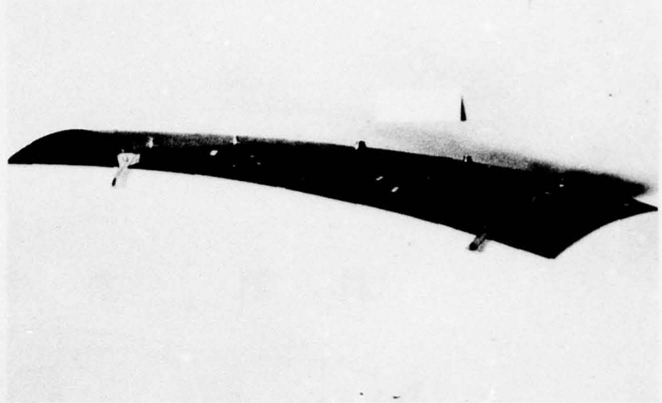
AS. X10
TRAPPES TRAIN PRINCIPAL
Fig 15



AS X10
AEROFREIN
Fig. 16

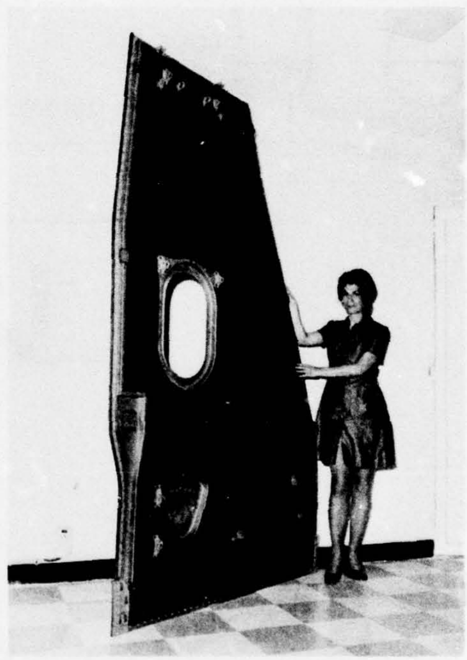


AS X10
FUSELAGE AVANT Fig. 17

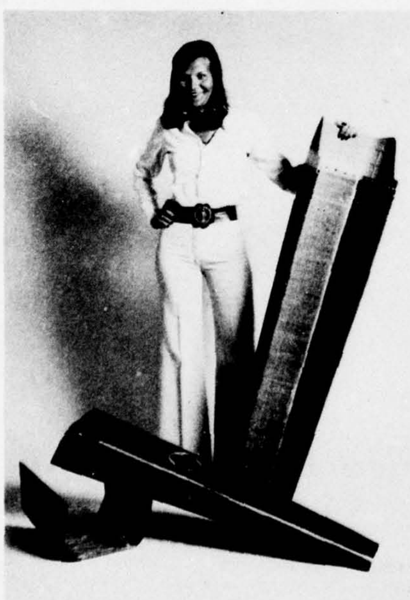


Porte de soute radio CORVETTE

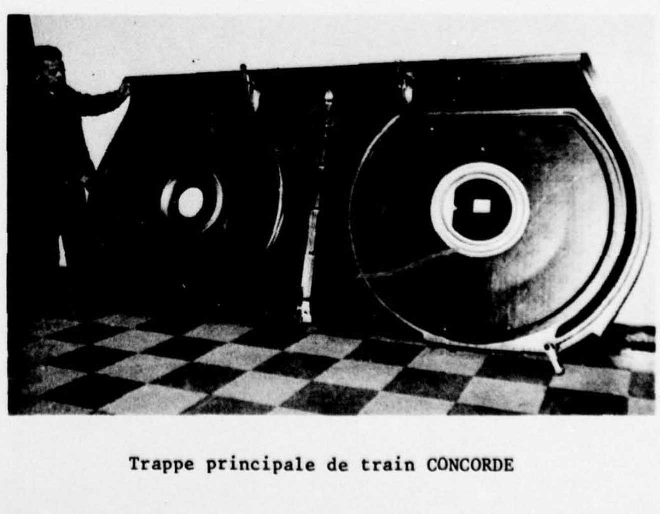
Porte de nacelle VAUTOUR



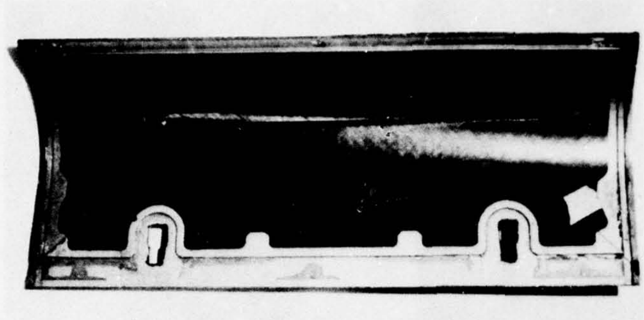
Trappe secondaire de train CONCORDE



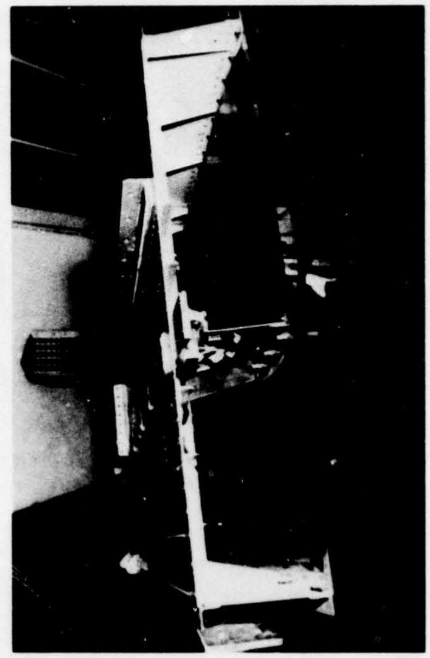
Carénages de servo-commande CONCORDE



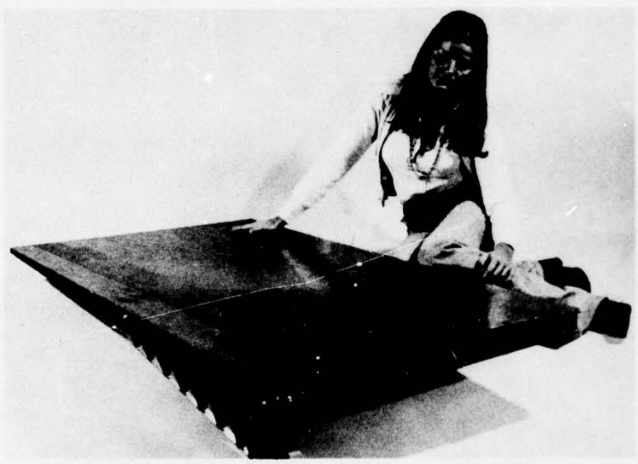
Trappe principale de train CONCORDE



Porte de visite Karman de voilure CONCORDE

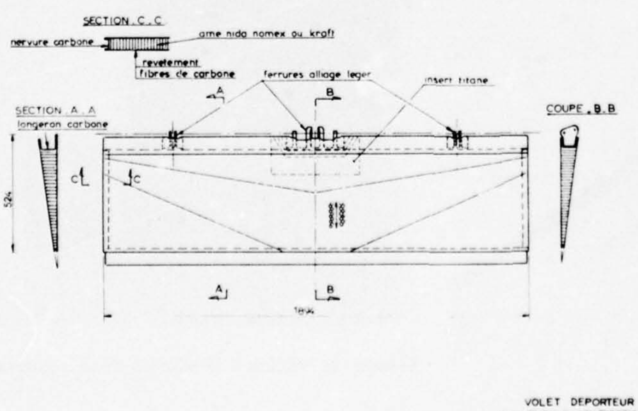


Structure Karman de voilure CONCORDE

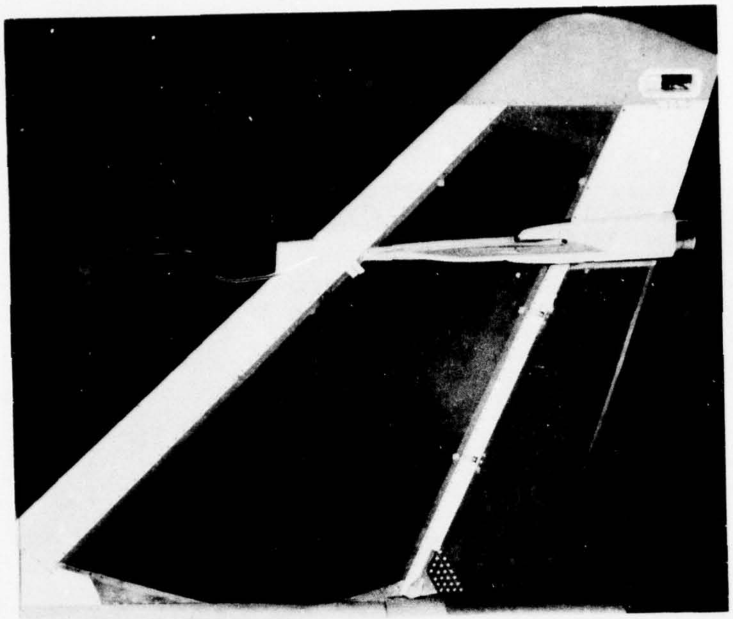


Elevons externes CONCORDE

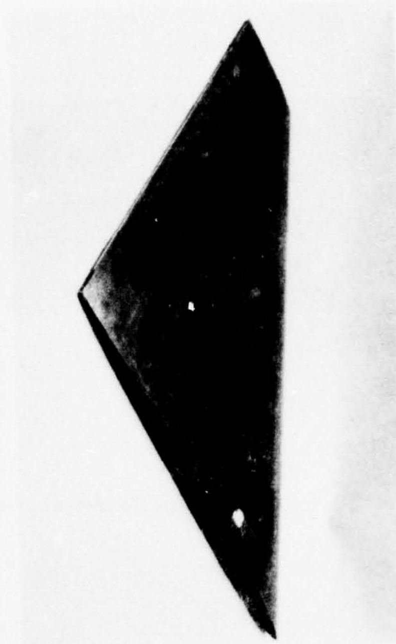
Elevons internes CONCORDE



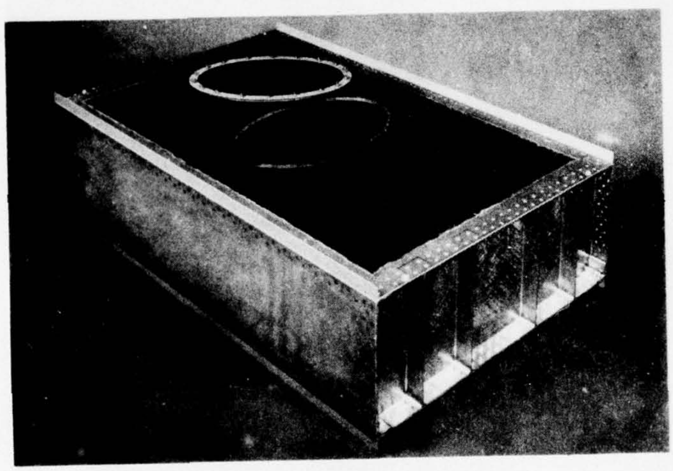
Lift dumper AIRBUS



Dérive MIRAGE 2000 pour les avions Marcel DASSAULT

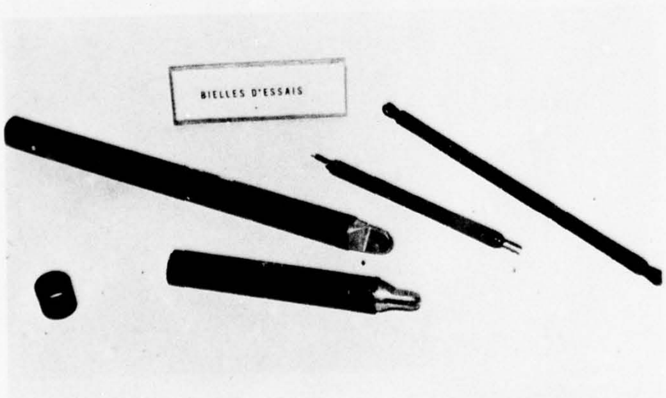


Voilure engin-cible



Panneau de revêtement voilure CONCORDE

Saumon de voilure CONCORDE Développement



Bielles

METAL TECHNOLOGY FOR FUTURE AIRCRAFT DESIGN

by
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1. INTRODUCTION

There have been many papers presented, and articles written, on the subject of metallic materials for future aerospace use by people far better qualified to discuss metallurgy than I. This paper will therefore look at the subject from a structural designers point of view to establish future needs and will also examine the way in which metallic materials stand up to the competition from their potential greatest rivals, the Graphite Composites. At the risk of stating the obvious let us first briefly remind ourselves of the needs of the finished article, the operational aircraft.

2. REQUIREMENTS

a. Strength and Stiffness.
The aircraft must meet the required flight envelope for both 'g' and speed.

b. Service Life.
Military aircraft are now expected to remain in Service for much longer periods and thus require a good fatigue life and resistance to the ravages of time, for periods of up to 20 years.

c. Vulnerability.
Damage Tolerance requirements are designed to combat the structure's susceptibility to manufacturing imperfections, and ensure that there is a reasonable tolerance to servicing and battle damage.

d. Repairability.
The probability is that the structure will suffer damage of some sort during its lifetime. It must be possible to take the necessary repair action without the need for complicated procedures.

e. Inspectability.
It must be possible to carry out successful inspections of the structure, using NDI equipment which can be applied rapidly and easily, even at squadron level.

f. Cost.
The structure must be cost effective both in terms of initial cost and in cost of ownership throughout its lifetime.

3. OPERATIONAL ENVIRONMENT

Figures 1 to 4 give a rather obvious look at some of the conditions under which the aircraft will actually be operated. These are:-

Fig. 1 Heat and Sand
Fig. 2 Salt Water
Fig. 3 The armourer's traditional favourite tool
Fig. 4 The universal adjuster

These are just a few reminders that we are building vehicles that will be used and not operated under laboratory conditions.

4. WHAT DO WE WANT?

4.1 Taking the requirements in turn we will first consider Strength and Stiffness.

Let us look, initially, at mechanical properties of materials as they might be applied to the wing of a future fighter aircraft. It will be convenient to take the wing designed in Carbon Fibre Composite as the baseline for comparison, and then see what properties would be required from the three main metallic materials in order to produce a wing of the same weight.

The wing to be considered is shown in Figure 5 and is designed for normal accelerations up to 9g and for speeds up to M = 1.9. The baseline carbon fibre wing is designed as a multi-spar structure on the grounds that this concentrates the bending material at the maximum distance from the neutral axis and provides maximum torsional rigidity. The weight penalty of the intermediate stabilising ribs is small because of their shallow depth. In the knowledge that the torsional stiffness of the wing was likely to prove a critical parameter in the design of the wing box a composite lay up of 1/3 low modulus Type 130 SC *CFRP at 0^o and 2/3 high modulus Type 200 SC CFRP* at $\pm 45^o$ was selected for the wing skins. A unidirectional longitudinal strength of 130,000 lb/in^2 was used for the 130 SC material and 94,000 lb/in^2 for the 200 SC material.

These figures make allowance for the expected statistical scatter in material properties and the likely operating temperature envelope of the aircraft.

* U.K. Specification

Let us now consider this same wing as though it were made from Aluminium Alloy, Titanium Alloy or Steel. It will be convenient to look at a number of chordwise sections at various span-wise positions along the wing, and show the thicknesses and material properties required for the same weight of wing skins. The results of these calculations are shown in Figures 6 – 9, with the required skin thicknesses for the baseline composite wing in the first column. The skin thicknesses for the other materials are derived merely from considerations of relative density for the same weight (I am assuming that we are unable to change material density as a parameter! if we could our problems would be over). Having thus derived a maximum skin thickness the usual parameters such as buckling, strength, torsional and flexural stiffness have been examined and required values derived for E, G, f_{ult} and f_n. Also, where necessary, adjustments have been made in spar spacing to obtain the required buckling strength and, where this produced impractical spacings at the outer end of the wing, consideration has been given to reverting to a skin-stringer form of construction.

In order to complete the picture a similar comparison has been made, Fig. 10 but with the Composite wing designed entirely in the high strength, low modulus, Type 130 SC fibre which seems to be more the current trend. In this instance the fibres had to be arranged with 3/4 fibre orientation at $\pm 45^{o}$ and ¼ at 0^{o} in order to obtain the necessary torsional stiffness.

What conclusions can be drawn from all this? In broad terms the increases required in the metal properties to compete with the baseline wing are shown in Fig. 11.

Also shown (Fig. 12) are the required increases based on the Type 130 SC Composite wing. From these results it can be seen that by far the greatest improvement required is in terms of Young's Modulus E, for a wing of this type which is very stiffness critical. It is of interest to note that the ultimate strengths of existing metallic materials would be adequate competition for the entirely high strength/low modulus composite wing but that there would still need to be substantial improvements in E. It is also of interest that, as the skin loading intensity decreases over the outer half of the wing, the metallic wings become more efficient as skin/stringer construction rather than multi-spar.

These effects are best illustrated (Fig. 13) by showing graphically the properties needed for an aluminium alloy for two criteria – strength and stiffness. The curves are plotted with a base that reflects the overall speed – manoeuvre needs of the aircraft in that it represents that proportion of a carbon fibre skin thickness that would be orientated for stiffness rather than strength. Thus, taking our baseline wing we have 2/3 fibres at $\pm 45^{o}$, and R therefore = 0.66. The wing comprised layers of both high and low modulus fibres. We can therefore see that, for stiffness, the aluminium alloy needs an E of 20×10^{6} lb/in^2 and f_{ult} of 95,000 lb/in^2. Note the large effects of using either all high modulus fibre or low modulus fibre.

These remarks apply to a very thin wing for a high speed aircraft, the emphasis would, of course, be different if the comparisons were to be made on fuselage structure. In the case of a fuselage it is probable that the emphasis on Young's Modulus (E) would be much less marked and the need for higher E and higher ultimate strength would be more evenly balanced.

4.2 Service Life

Most new fighter aircraft will be required to have a fatigue life of at least 6000 hours, perhaps as long as 10,000 hours. In order to achieve such a life tensile stress levels must be limited to a value below that required to achieve the normal static safety margin. How critical is this in a fighter aircraft, where the structural design is dictated by so many other factors, and how much value can therefore be attached to improved fatigue performance of materials? It is very difficult to put exact numbers to this but some broad assumptions can give a feel for the trend to be expected in structure weight from improved fatigue resistance. Fig. 14 shows the relationship between stress per 'g' and life for the H.S. Hawk, which was designed to a very severe fatigue spectrum. If we assume that 8% of the structure is sized on the basis of fatigue considerations we can see the effect on total structure weight of lifting this curve, i.e. increasing the S in the S–N curve. If the material performance in fatigue could be improved by, say, 5% the allowable stress/g increases from 1520 lb/in^2 to 1600 lb/in^2 and, obviously enough, the weight of the fatigue critical parts should be reduced by 5%. This would mean a total weight saving on the airframe of 0.4%, or 14 lb. on an 3500 lb. structure. Similarly a 25% increase in material fatigue strength would mean a total weight saving of 2%, or 70 lb. on an 3500 lb. structure.

Against this it is suggested that the fatigue performance of carbon fibre composite is extremely good. I am not sure that we have heard the last on this score and, whilst the statement may be true for the fibres themselves, it is quite possible to have "fatigue" failures due to the breakdown of the resin system if the detail design is such as to induce high local inter-laminar shears.

It would thus appear that, whilst obviously any improvements in fatigue strength of metallic materials would be welcomed, large increases are needed to give any appreciable weight savings for a given life, and even larger increases to match the performance of the composites.

However, there is more to service life than just fatigue strength. There is the ability to withstand the accumulation of years of minor damage caused by servicing, hail, heat, moisture – and it must be said that the performance of the composites remains to be fully proven in this respect. Thin laminates are particularly prone to damage of this sort, even if backed up by honeycomb cores – we have had an instance (Fig. 15) where a dropped block of wood produced a very unacceptable hole in a component we were about to test! It therefore remains to be seen how components of this sort will survive in a service environment and whether, in the end, they will supplant the use of metals.

4.3 Vulnerability

There is now a much greater emphasis on damage tolerance as an aid to improved structural in-service performance. There are two basic objectives of damage tolerance criteria:-

1) Safety
2) Economic and operational viability.

Safety is usually defined in terms of a required residual strength following the propagation of specified initial defects during the nominated inspection period.

Economic and operational viability are defined in terms of an acceptably long period of service usage between the acquisition of a structure and the time when repairs become uneconomic. Also there must be an acceptably long interval between periods of inspection and between the application of necessary repairs.

The whole concept is based on the assumption that an initial flaw will always be present somewhere in the structure, usually in the very last place you considered possible! We should bear in mind, also, the possibility of service damage as illustrated earlier. A very much simplified approach to the subject is shown in Fig. 16, which shows the relationship between flaw size, type and level of inspection period, crack propagation rate and residual strength. The factor in the crack growth column can be varied depending on the ability of the structure to satisfy "fail-safe" criteria; for a fully "fail-safe" structure this factor may approach unity.

There is therefore a direct relationship between material properties – initial flaw size and crack propagation rate – and the maintenance philosophy of the aircraft in terms of over haul and inspection periods. There is also the inevitable compromise to be achieved between allowable stress levels, and hence weight, and service usage. The ideal service aircraft, which requires no inspection or maintenance throughout its life, is obviously the heaviest.

These criteria are as important as, if not more so, than the traditional fatigue life approach. In order to increase fracture toughness and resistance to crack propagation we need materials with very low impurity levels and with very narrow limits on element composition to achieve the greatest consistency and the best possible ratio between fracture toughness and material tensile properties.

This is best illustrated by looking at the current and immediate future aluminium alloys, Fig. 17. This illustrates the trade-off between static tensile properties and the ability to resist Exfoliation and Stress Corrosion Cracking. Reductions in tensile properties of approximately 12% are necessary to ensure freedom from these problems. The table also illustrates the effect of increased purity and close alloying limits on fracture toughness K_{IC}. It will be seen that increased purity provides a very considerable increase in this parameter which is still further enhanced by more closely controlled alloying of the material.

It should be noted that there is now a considerable back-log of experience and test evidence available for metals on the subject of vulnerability whereas much remains to be done in the field of composites although preliminary results look promising. As seen very thin composite skins, in particular, are prone to damage from the traditional tools mentioned earlier.

4.4 Repairability

The properties required to make for easy structural repair are similar to those that influence the basic material choice in the first instance. In general both metallic and composite structures will be repaired by bolt-on patches of some sort, these may be external patches for field type repairs or more complex flush patches for maintenance base repairs. This is an area where metallic materials may prove to be better than composites since it is possible to take advantage of interference fit fasteners for long term repairs whereas the ideal solution for composites, the use of a hot glueing technique, may not be practical. In fact, it may be argued that considerations of the ability to repair may limit the allowable stresses in a composite component to some value below the optimum.

Required properties for repairability are therefore a good tolerance to crack propagation, good fatigue strength for the bolted joint, the ability to clean up the damaged area easily and to obtain a good finish before patching, and the ability to ensure through NDI that all damage has indeed been removed. This may be particularly difficult with composite structures where there is a tendancy for resin failure to spread along the fibre orientation and to propagate through the thickness. It is therefore likely to be difficult to ensure that you have got back to clean virgin structure before repairing.

4.5 Inspectability

However well designed an aircraft structure may be in the first instance it is almost inevitable that it will suffer some sort of problem, whether from fatigue or corrosion, during its lifetime. As has already been shown there must be a direct link between the structure's susceptability to damage and the method of organising the servicing and overhaul periods. The setting of these periods depends on the design and the material chosen, but the time allowed for Inspection, and the total cost of doing the job, must depend on the ease with which the inspection can be carried out. In general homogeneous materials, such as metals, fail by surface cracks that propagate into the material. Exceptions are failures from internal material defects, a rare occurrence, or from welds. Composites are more likely to fail from internal interlaminar defects which could reduce the strength of the laminate without being immediately obvious.

There is therefore an economic advantage if cracks can be allowed to develop to a size where they are readily detectable before becoming critical to the structure. Alternatively the means of inspection to detect smaller cracks must be cheap and simple to operate. For metallic materials there is a wide range of well established techniques for doing this but for composites there will be a need for instrument inspection procedures, such as C–Scan, holography etc., which will require expensive equipment which is time consuming to operate and difficult to interpret. It would therefore seem, for the moment at any rate, that metallic materials have an advantage in the field of inspectability.

4.6 Cost

This is perhaps one of the most important features to be considered if metallic materials are to hold their own in the future. If we look at the material utilisation factors being achieved at the moment we see that the composites factor is approximately 1.2 to 1.3 and metals is approximately 7.0, i.e. we get more lb. of finished structure per lb. of composite than we do with metals. However the cost of fabricating parts in composites is currently involving many more man hours of labour than numerically machined metal parts, or even fabrication from sheet metal. In consequence the current cost of a Composite component is roughly the same as that of a sheet metal fabricated part, and is therefore expensive compared with N/C machining.

If we are to make the most effective use of metals in the future we need to make the maximum use of the material that we buy. This means careful attention to detail design to minimise the required billet size if the part is to be machined from bar or plate, and the use of more forgings or castings which require only small amounts of metal removal. Machining technology is continuously improving and the use of modern N/C machines makes it practicable to fabricate complex parts in this way which would have been impossible a few years ago, for instance the cost of machining Titanium parts has been reduced by 30–50% in recent years with further reductions still to come. It is also possible to conserve on material by designing parts which utilise the weldability of 6A1–4V Titanium, and some of the better corrosion resisting steels, Electron Beam welding is a particularly useful technique here.

For the future, the use of Isothermal forging of Titanium alloys is already showing reduction in material utilisation factors from around 7 to 2–3 compared with conventional forgings. For smaller parts the research work on Powder Metallurgy may well prove to be the most cost effective solution in the long term, and diffusion bonding for smaller section members could produce economic skin-stringer combinations which would save the scrap associated with machine skins. Another process which is showing great promise takes advantage of the properties of Titanium to form parts by super-plasticity. Many small components are now being made using this process and its use is being extended to larger parts.

Incidentally, I am sure we have all seen comparisons of weight, or cost, of a part to be fabricated in the new wonder material (usually composites!) compared with an existing part. I think we have to be careful that the comparison is not between apples and oranges, in many instances the existing part could, with today's knowledge, be designed in a more economical fashion which would make the comparison less odious – like statistics, trade-off studies can very often be pointed in the direction the trader wants to go!

4.7 Let us then summarise What do we want for future fighter aircraft?

a) **Material Properites**

In order of merit:-

- Improvement in E highly desirable.
- Improved Fracture Toughness and resistance to Crack Propagation.
- Improved Fatigure Strength.
- Higher static strength properties.
- Improved Resistance to Exfoliation and Stress Corrosion.

b) **Utilisation**

In order of merit:-

- More efficient forgings.
- More efficient castings.
- Improved welding.
- Powder Metallurgy †

† This, I think, is probably the correct position for Powder Metallurgy from a purely fabrication point of view, but in terms of its importance to the development of the metallurgy of new materials it must assume a position of very much greater prominance.

5. WHAT HAVE WE GOT?

Having looked at the desirable let us now look at the attainable, or at least as far as we can see into the immediate future.

5.1 *Aluminium Alloy*

The highest strength alloys are based on an aluminium – zinc combination but this combination requires the addition of elements to refine the grain size. Manganese or Chromium have been used for this purpose but the material needs fast quenching with the consequent dangers of high internal stresses. As a result the performance in terms of stress corrosion has been poor. In the newer materials the choice of zirconium as a re-crystallisation inhibitor produces a low quench sensitivity which allows high strengths to be maintained in thick sections and the use of low quench rates following solution heat treatment with a consequent improvement in the stress corrosion characteristics. The addition of copper as an alloying element provides precipitation to give the required properties after heat treatment, and the susceptibility to exfoliation is reduced by over-aging beyond the point where peak hardness is obtained, e.g. T76 or T73 temper. Increased fracture toughness and resistance to crack growth are provided by the low impurity levels.

Further improvements are being sought in the family of Aluminium - Copper alloys with great emphasis being placed on the attainment of super purity provided by the low constituent impurity levels combined with a better understanding of the working process during manufacture. However, the improvements already discussed in the Aluminium - Zinc alloys, with their higher strengths and fatigue performance approaching the Al-Cu alloys, must make them the prime choice for future fighter aircraft.

We also hear of the use of an Aluminium - Magnesium - Lithium alloy by the Russians. This is reputed to have a useful increase in stiffness and to have saved 15% of the weight of a wing top skin.

5.2 *Titanium Alloys*

For any aircraft to be designed within the immediate future the prime choice of Titanium alloy must still be the 6Al –4V variant. This has now been in use for a number of years and has proved itself in many aircraft in many forms, sheet, bar, forgings and, more recently, the development of castings with excellent properties approaching those of the wrought material. A variation in the heat treatment, – beta annealing, improved the fracture toughness without significant loss of other properties.

Many other alloy compositions have been studied but for structural applications, only one – "Corona-5" – has shown really significantly improved properties. This alloy, sponsored by U.S. Naval Air Systems Command, has tensile properties only slightly better than Ti–6A1–4V, but its fracture toughness is at least twice as good. A number of alloys have been developed with higher strength than 6–4, e.g. Beta-C, and "6–22–22", but none have gone into quantity production, or achieved as good a balance of properties as 6–4. There are also improvements being sought in the IMI range of materials where the 550 alloy is being developed to improve fracture toughness.

5.3 Steels

There is not a great deal of movement on the steel situation, there are already available some excellent materials such as 300M, which has very high strength with a moderate – but adequate – fracture toughness, D6AC which trades some of the tensile properties for better fracture toughness and the HY 180 material which has outstanding values of K_{IC} combined with good tensile properties. In view of the current thoughts on vulnerability Fig. 18 may be of interest, this shows comparative critical crack lengths for a number of different materials. The two new materials, Corona-5 and HY 180, are still in the development stage.

5.4 Unconventional materials

During solidification and slow cooling of a cast billet the alloy structure can become far from optimum as regards segregation, grain size and directionality, size and distribution of constituent particles etc. Reheating and working of the ingot then becomes necessary to improve these characteristics, the processing to improve one property is frequently detrimental to another and the final material properties represent a compromise depending on the emphasis required. It would obviously be advantageous to make alloys that did not possess these problems in the first place and this can be done in some instances by such processes as powder metallurgy or spray casting.

In particular Spray Casting has been used to produce Aluminium alloys with higher tensile properties and moduli than has been possible by the normal melting process.

Both the Spray Casting technique for Aluminium alloys, and Powder Metallurgy for Aluminium, Titanium and Steels, give the metallurgist the opportunity of making element combinations that would not otherwise be possible. Although the products of these processes are still in the development stage it appears that improvements in room and elevated temperature properties and elastic modulus are possible to a greater extent than would be possible by conventional alloying and we look forward to having some production sized billets to play with.

Another technique under investigation is the use of laminated metals. This can be achieved either by rolling an ingot which is made from layers of the required material – the rolling process welding together the different materials – or by casting an ingot which contains the required distribution of different materials. I am told that, with the latter process, it is possible for the structural engineer to have his name permanently embedded in the material, rather like a stick of rock – it is perhaps open to doubt how many would take advantage of this possibility!

There still remains, of course, the possible use of fibre reinforced metal matrices. In addition to having increased properties these would be able to resist, more easily, the type of operational damage discussed earlier and would also be able to withstand higher operating temperatures. The fibres could be continuous, as with CFRP, or in the form of discrete whiskers or elements which could be positioned to suit the requirements of the component. Laboratory type specimens have shown extremely high moduli – one of the prime requirements we have established – with tensile properties approaching those of the Titanium alloys. Ductility, however, is low at around 1–2 per cent.

With all new materials the material manufacturers are less and less willing to provide the very high funding necessary to finance such ventures and even Government agencies must assess the relative priorities. It may therefore remain to be seen whether some of the promising possibilities ever become a production material.

CONCLUSIONS

How then do metallic materials stand up to the competition from composites and what is the possibility of achieving the improvements required? It has to be admitted that the increase in Young's Modulus (E) needed are highly unlikely to be achieved and that, therefore, metallic materials face very strong opposition for a thin fighter wing where stiffness is the prime consideration – particularly if high modulus fibres are used. It also has to be admitted that the competition will be intense in fuselage structure particularly if low cost fabrication from composites becomes a reality.

However I would not dream of suggesting that all Metallurgists should hand in their resignations and take up brick laying (Fig. 19). There will be many parts of the future fighter aircraft that will be made from metallic materials, in particular areas of intense local high loading, sub-structures, areas experiencing high temperatures, areas where the inherent toughness and ability to withstand damage of metals are needed. Structural designers are a conservative lot (Fig. 20) and there are still many unknowns regarding long term durability of composites, allowances of 15% are being quoted to cater for these unknowns. There is also a great possibility that an over cautious airworthiness approach will still further whittle away the composite advantages. But there is no room for complacency, the maximum possible effort must be given to improving metal utilisation, reducing costs, improving properties – if this is done I am sure that there is still a great future for metals in fighter aircraft.

FIG. 1 HEAT AND SAND

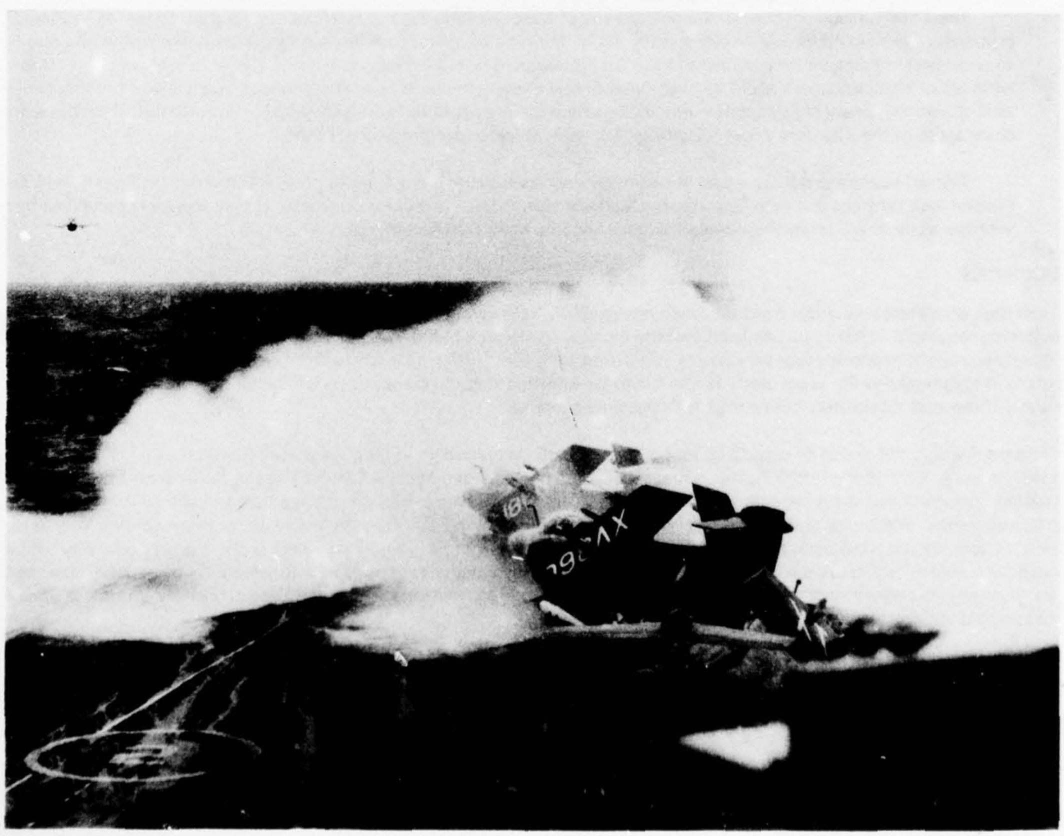

FIG. 2 NAVAL ENVIRONMENT

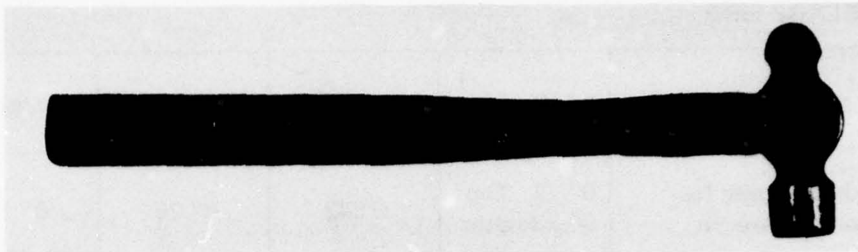

FIG. 3 THE ARMOURERS FAVOURITE TOOL

FIG. 4 THE UNIVERSAL ADJUSTER

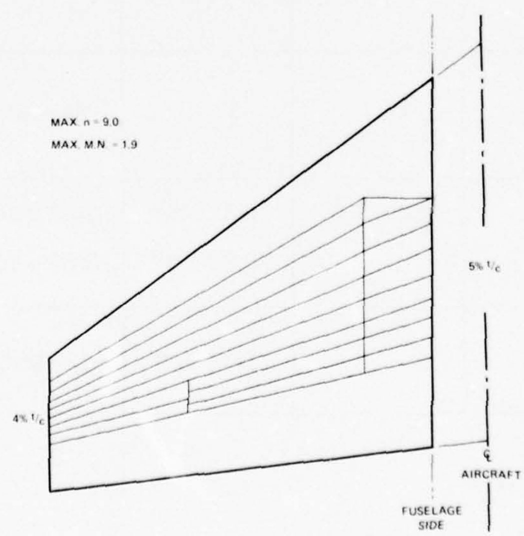

FIG. 5 WING GEOMETRY

AT FUSELAGE SIDE

MATERIAL	COMPOSITE †	AL. ALLOY	TITANIUM	STEEL
Skin thickness for constant weight	0″.72 Top 0″.735 Btm.	0″.42	0″.26	0″.13
Spar spacing or skin/stringer	9″.2	9″.2	6″.6	3″.3
G Required lb/ins^2 E Required lb/ins^2	4.5×10^6	7.2×10^6 17.9×10^6	11.3×10^6 28.3×10^6	21.4×10^6 53.6×10^6
f Required lb/ins^2	f_{ult}=50 000	f_{ult}=80 000	f_n=120 000	f_n=233 000

† $^2/_3$ High modulus at $\pm$ 45^o

$^1/_3$ High strength at 0^o

FIG. 6 SECTION AT FUSELAGE SIDE (Mixed Composites)

40% SEMI–SPAN

MATERIAL	COMPOSITE †	AL. ALLOY	TITANIUM	STEEL
Skin thickness for constant weight	0″.36	0″.19	0″.105	0″.059
Spar spacing or skin/stringer	8″.1	5″.1	Skin/Str.	Skin/Str.
G Required lb/ins^2 E Required lb/ins^2	4.5×10^6	8.1×10^6 20.3×10^6	14.2×10^6 35.4×10^6	24.9×10^6 62.4×10^6
f Required lb/ins^2		f_n=79 200	f_{ult}=150 000	f_{ult}=265 000

† $^2/_3$ High modulus at $\pm$ 45^0

$^1/_3$ High strength at 0^o

FIG. 7 SECTION AT 40% SEMI-SPAN (Mixed Composites)

66% SEMI–SPAN

MATERIAL	COMPOSITE †	AL. ALLOY	TITANIUM	STEEL
Skin thickness for constant weight	0".18	0".089	0".056	0".032
Spar spacing or skin/stringer	6".27	Skin/Str.	Skin/Str.	Skin/Str.
G Required lb/ins^2 E Required lb/ins^2	4.5×10^6	8.7×10^6 21.7×10^6	13.6×10^6 34.0×10^6	24.0×10^6 60.0×10^6
f Required lb/ins^2		f_{ult}=78 000	f_{ult}=125 000	f_{ult}=220 000

† $^2/_3$ High modulus at $\pm 45^o$

$^1/_3$ High strength at 0^o

FIG. 8 SECTION AT 66% SEMI-SPAN (Mixed Composites)

80% SEMI–SPAN

MATERIAL	COMPOSITE †	AL. ALLOY	TITANIUM	STEEL
Skin thickness for constant weight	0".135 Top 0".105 Btm.	0".064	0".04	0".02
Spar spacing or skin/stringer	5".34	Skin/Str.	Skin/Str.	Skin/Str.
G Required lb/ins^2 E Required lb/ins^2	4.5×10^6	8.2×10^6 20.4×10^6	12.9×10^6 32.2×10^6	23.2×10^6 58.0×10^6
f Required lb/ins^2		f_{ult}=50 000	f_{ult}=80 000	f_{ult}=145 000

† $^2/_3$ High modulus at $\pm 45^o$

$^1/_3$ High strength at 0^o

FIG. 9 SECTION AT 80% SEMI-SPAN (Mixed Composites)

AT FUSELAGE SIDE

MATERIAL	COMPOSITE †	AL. ALLOY	TITANIUM	STEEL
Skin thickness for constant weight	1".06	0".59	0".37	0".21
Spar spacing or skin/stringer	9".19	9".19	9".19	9".19
G Required lb/ins^2 E Required lb/ins^2	3.5×10^6	5.5×10^6 13.6×10^6	8.2×10^6 20.4×10^6	13.9×10^6 34.9×10^6
f Required	f_{ult}=38 000	f_{ult}=60 000	f_{ult}=89 250	f_n=154 000

† ¾ High strength at ± 45°

¼ High strength at 0°

FIG. 10 SECTION AT FUSELAGE SIDE (All high strength)

INCREASES TO MATCH BASELINE WING

MATERIAL	AL. ALLOY	Ti ALLOY	STEEL
E	95%	112%	108%
f_{ult}	34%	2%	74%

FIG. 11 METAL IMPROVEMENTS REQUIRED (MIXED COMPOSITE TYPES)

INCREASES TO MATCH ALL HIGH STRENGTH COMPOSITE

MATERIAL	AL. ALLOY	Ti ALLOY	STEEL
E	48%	54%	44%
f_{ult}	✓	✓	21%

FIG. 12 METAL IMPROVEMENTS REQUIRED (ALL HIGH STRENGTH COMPOSITES)

ALUMINIUM ALLOY PROPERTIES REQUIRED TO MATCH COMPOSITE LAMINATES

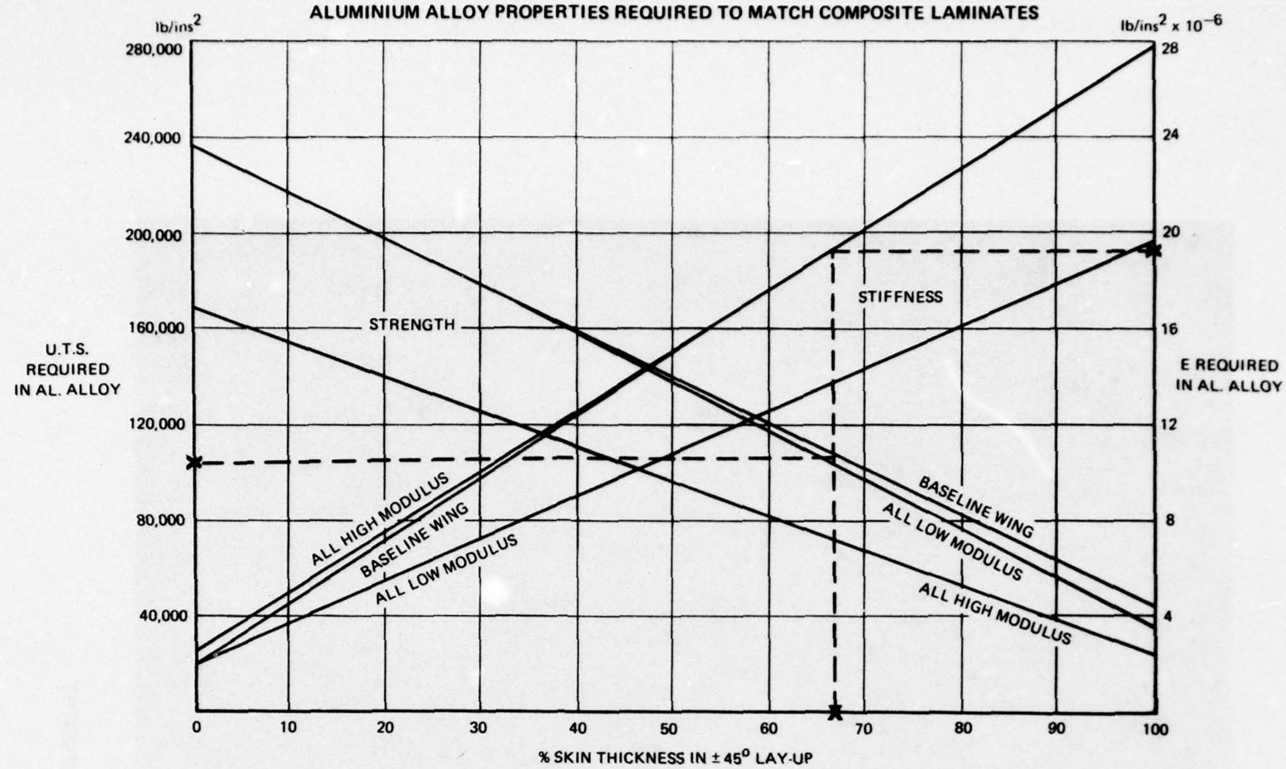

FIG. 13 ALUMINIUM ALLOY IMPROVEMENTS REQUIRED FOR STRENGTH OR STIFFNESS

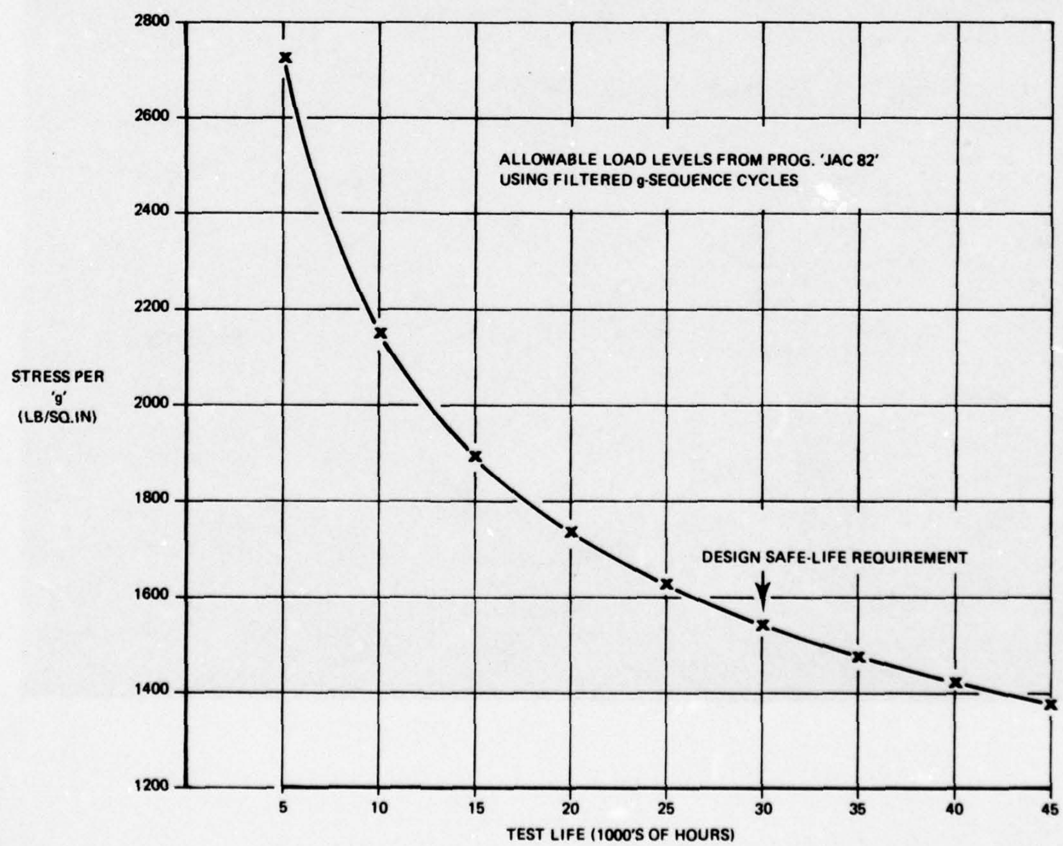

FIG. 14 HAWK ALLOWABLE STRESS PER 'G'

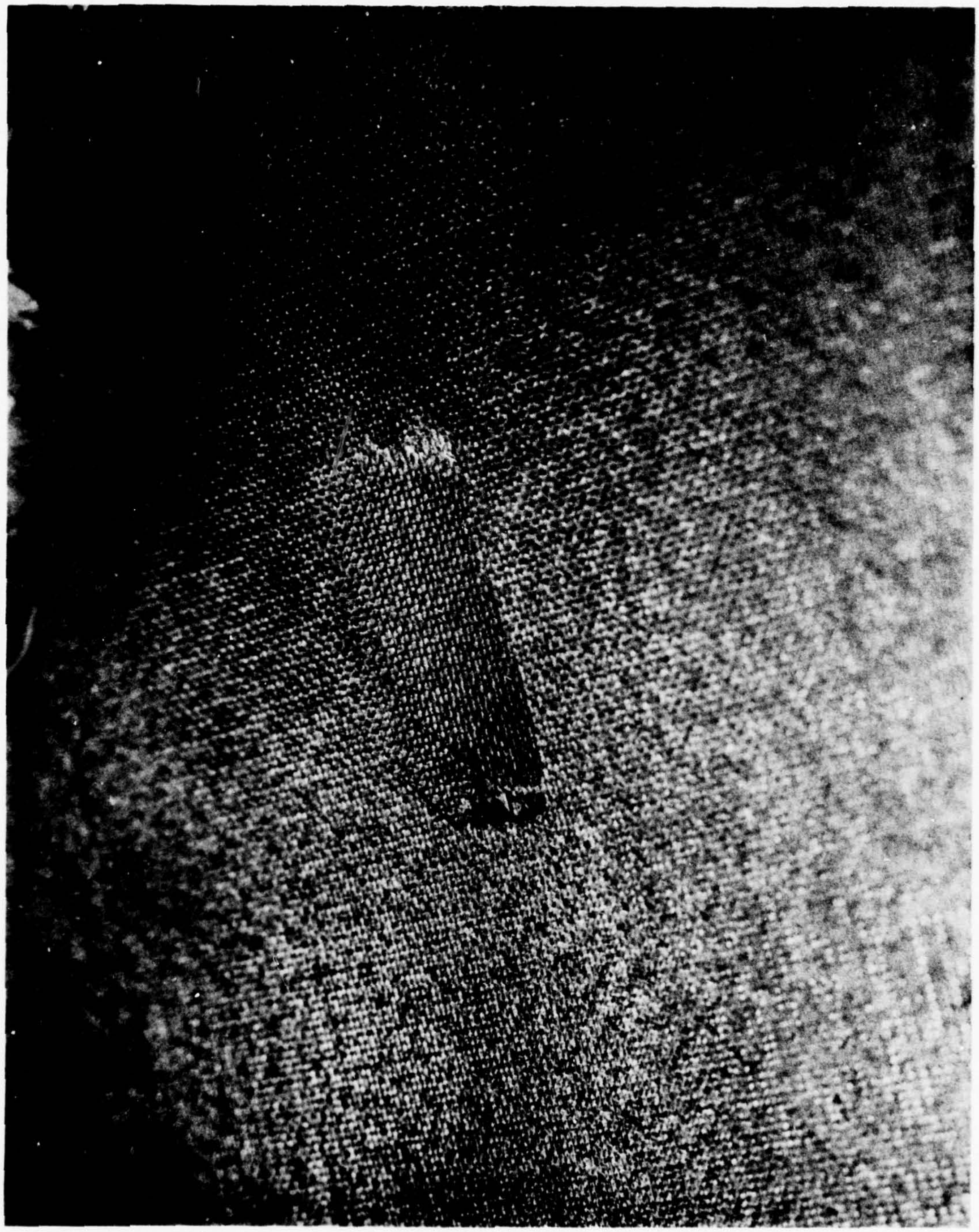

FIG. 15 DAMAGED COMPOSITE COMPONENT

Selected inspection standard	Detection capability (min. flaw size)	Inspection interval	Damage Tolerance Requirement			Comments
			Initial flaw	Growth period	Residual strength	
(1) Manufacturer's inspection. Service non-inspectable.	a_{01} (ex-factory) [0.05"]		a_{01} [0.05"]	F_1 x service life [2.0]	P_1	Severe requirement to cover "blind" situation. Combines "safe life" and "damage tolerance" approaches. Penalises "uninspectable" design.
(2) Depot level (M.U.) major overhaul. N.D.I. with components removed.	a_{02} ($>a_{01}$) [0.05"]	η_1 hours	a_{02} [0.05"]	$\kappa_1 \eta_1$ [2.0]	P_2 ($<P_1$)	Possible weight savings and reduced service problems, in return for scheduled withdrawals from service and the cost of structural repairs.
(3) Depot or base (airfield) overhaul. N.D.I. with component "in situ".	a_{03} ($>a_{02}$) [0.25"]	η_2 hours ($\leqslant \eta_1$)	a_{03} [0.25"]	$\kappa_2 \eta_2$ [2.0]	P_2	Less sensitive inspection than in (2) but feasible to apply in service. May still permit weight savings c.f. (1).
(4) Base level (airfield) close visual inspection.	a_{04} ($\gg a_{03}$) [2.0"]	η_3 hours (or flights)	a_{04} [2.0"]	$\kappa_3 \eta_3$ [2.0]	P_2	Much coarser level of inspection than (2) or (3). Applicable to flight safety clearance with low values of $\kappa_3 \eta_3$.

Increasing flaw size ↓

Decreasing inspection period ↓

[ ] Values in Mil. Spec.

<u>Suggested Damage Tolerance Requirements</u>

<u>Slow Crack Growth Structure</u>

FIG. 16 DAMAGE TOLERANCE APPROACH

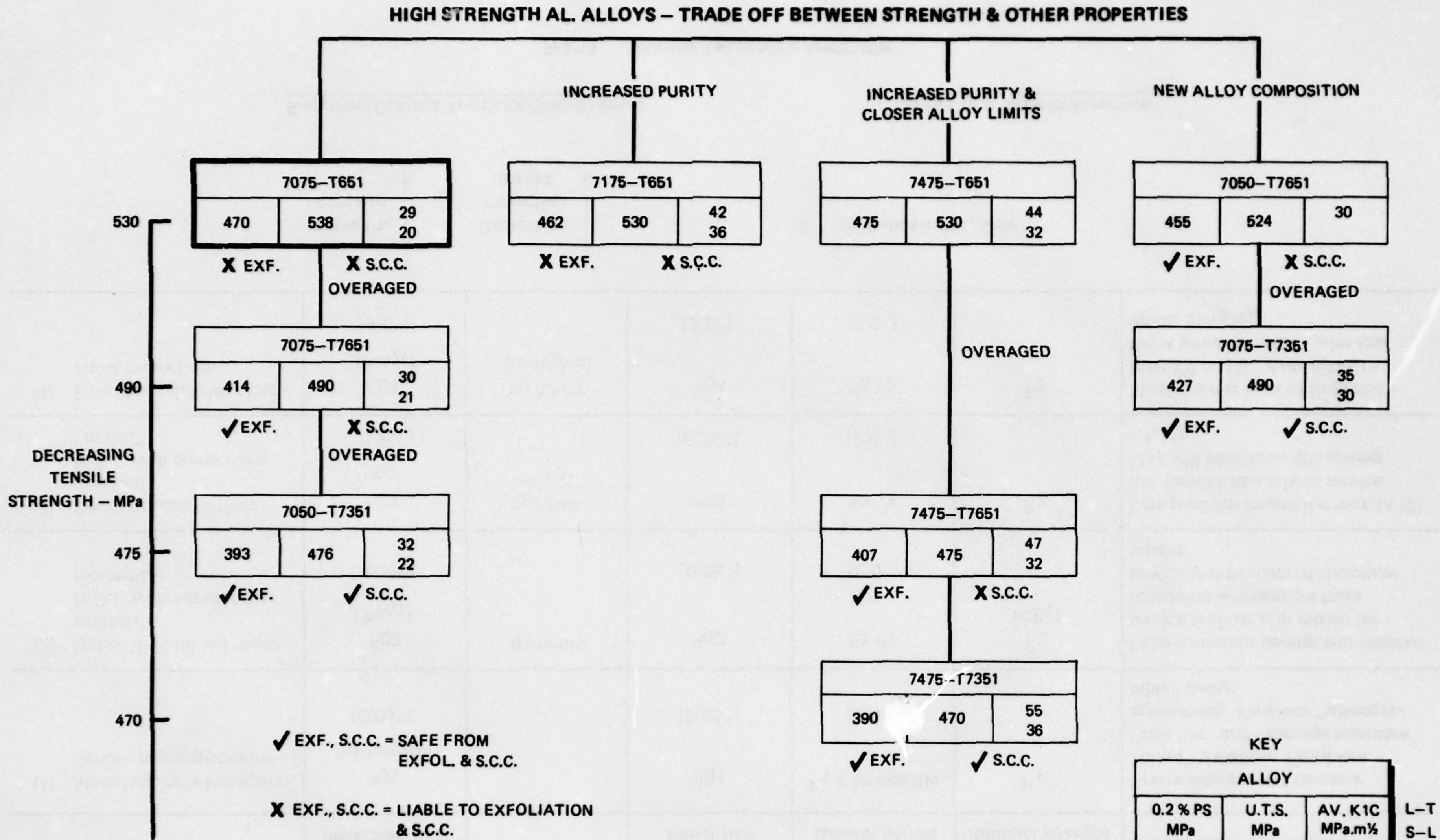

FIG. 17 MATERIAL TRADE OFFS

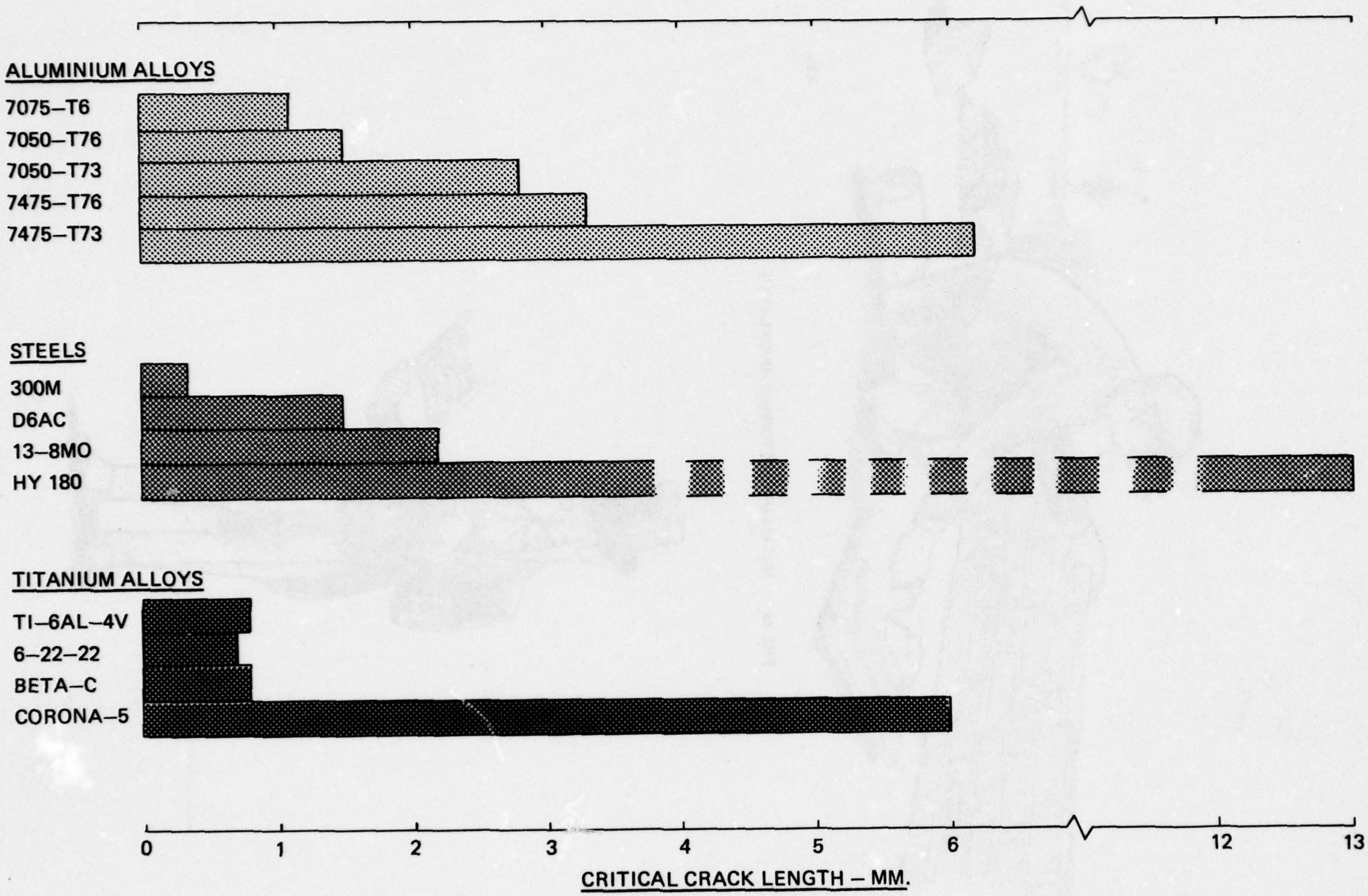

FIG. 18 CRITICAL CRACK LENGTHS FOR A1, Ti ALLOYS AND STEELS

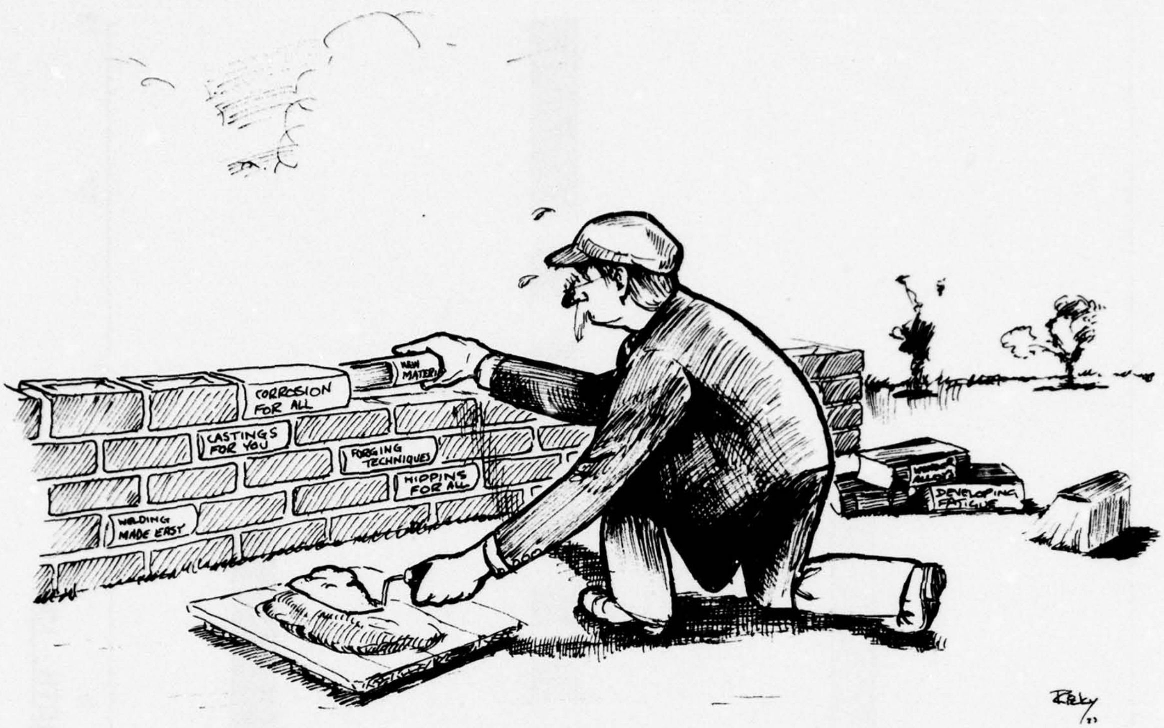

FIG. 19 METALLURGIST TURNED BRICKLAYER

FIG. 20 STRUCTURAL ENGINEERS ARE A CONSERVATIVE LOT

DISPLAY SYSTEMS AND COCKPIT DESIGN

by

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Summary

In this paper a concept for a new cockpit design technology is presented.
The need for a new technology is dictated by the technical requirement for saving cockpit weight and space, resulting from the high g-operations requirement, - and from the operational requirement for limiting the quantitative personnel requirements.
The basis for the new technology is given by the present knowledge concerning the information processing capabilities of man, and by the empirical results of measurements of the time-budget of man in man-machine operations.
From this the display and control modalities are derived, which ensure better utilization of the pilot's capabilities) in the future fighter aircraft. This concept is considered to be the only chance for the future, to increase the system capacity of manned fighter aircraft, compared with the equipment-overloaded aircraft of today.
The paper ends with the principle allocation of the display- and control-function to the equipment and with an example for the required characteristic for the functions in accordance to the time budget considerations.

1. Basic considerations concerning aircraft cockpit design

From an aircraft designer's point of view it is desirable to have a single seat aircraft with a cockpit design that allows the one operator to perform all the tasks necessary for a successful mission sortie under the condition of an acceptable operator's workload.

In this context it has to be called to mind that the design of twin seat aircrafts was not a "free decision" but was caused by the necessity of housing a large number of equipment and of not overloading the single operator. The fact that the role of the second crew member was somewhat ambiguous from the very beginning, can already be concluded from the "label" given to him. He is called "system operator", "navigator", "Kampfbeobachter" (combat-observer). None of these names encompasses his role completely, only all of them together would do it.

Table 1 presents a survey of general criteria for comparison of the single seat vs. the two seat cockpit. The two seat cockpit was favoured for the present fighter aircrafts because of its superiority in the three following criteria:

Evaluation criteria	Cockpit configuration: single seat	assessment	two seat
	qualitative	quantitative	qualitative
1 Weight required	+	<	–
2 Space required	+	<	–
3 Number of systems/equipment installable	–	<	+
4 Number of equipment needed per A/C	+	<	–
5 Number of crew functions performable simultaneously	–	<	+
6 Number of crew functions performable per time intervall	–	<	+
7 Number of crew personnel required	+	<	–
8 Number of maintenance personnel required	+	<	–
	5+ 3–		3+ 5–

Table 1: Cockpit configuration assessment criteria

- number of *systems/equipment installable*
- number of functions simultaneously performable by the crew
- number of functions performable per time interval by the crew

This superiority was based on the technology available, which did *not* allow a *significant*
- reduction of the number of functions to be simultaneously performed by the crew
- reduction of the number of functions to be performed per time interval by the crew.

So the goal is to realize these reductions for which the quality of achievement depends to a great extent on the state of the art in technology and on the adherence to human engineering design criteria which have a regulative function in using advanced technologies for a manned vehicle. The complete changeover in cockpit design *presently* observed is caused (1) by the tactical needs e.g. high-g-manoeuvres, and (2) by the advanced development of the electronic technique, which induces new requirements particularly in the areas of threat information, friend/foe identification and ECM measures, resulting in a new approach to cockpit design.

The internal geometry of a single seat cockpit in accordance to today's technical requirements and to the possibility of the present and future electronic technique will show more or less the picture as presented in figure 1, with the following equipment characteristics:
- head-up display as primary flight operation display
- two TV/tabular displays (to the right and left), mounted close to and in the shadow of the anti-glare shield for aircraft systems operation, mission systems operation and "forced message" (including warning) operation
- a horizontal display in the middle part
- possibly a reclinable ejection seat
- pedals
- throttle box
- single row left and right hand console

The cockpit of the F18, is - as far as we know - the *first* cockpit of this new generation of fighter aircraft. And it shows this typical layout described above. This layout is mainly determined by the advanced technology available.

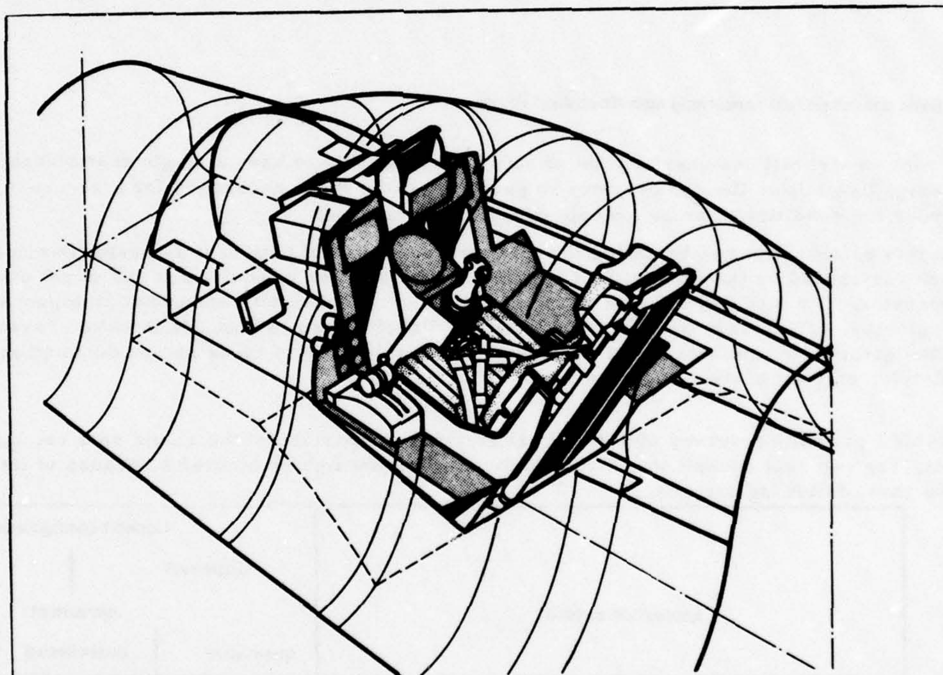

Figure 1: General view of cockpit layout for the future fighter aircraft

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Besides that technological approach we need a methodological basis, which is applicable to all future projects of designing a cockpit according to its operational mission and capability requirements. And this methodological basis must provide the following:

Make optimum use of man's channel capacity (perception, information processing and handling activity capabilities), within the cockpit design to meet the cockpit system capacity required for the future fighter aircraft.
What does that mean, "make optimum use of man's channel capacity" by design?
It means "automation", "integration" and "multi-purpose" of the cockpit equipment.
The TORNADO represents a first step in this direction, but still a two men crew is needed. The F 18 represents an example leading the way to integration by automation, using multi-purpose equipment.

But we need to know that automation as such, and application of advanced electronic technologies per se do not render the effect desired. By a simple example it can be shown that systems of nearly equal capabilities in electronic data processing impose extremely different load on man's time budget. The example in table 2 shows time consumption for manual input sequence (standard message) by three different Air Traffic Control (ATC) systems.

"LH 1423 CLEARED TO FL 210"

	duration in sec.
Mediator	4.4 - 6.5
NAS	5.1 - 7.8
GATC - 80	2.0 - 2.8

Table 2: **Time consumption for input sequence**

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This example shows that the mere application of electronic data processing does not automatically lead to a reduction of man's workload within the system. On the contrary, it could lead to an increase of workload and therefore to reduction in system capacity.
Therefore it is the objective of this paper to present the methodological concept for making better use of the operator's total channel capacity, by designing the cockpit in accordance with man's time budget for weapon system operation.

2. The time-budget of the human operator

The time budget of the human operator is defined as:
The total time needed by the operator (in accordance with his channel capacity) to perform a given task within a given time interval.

The notion "channel capacity" comprehends the human operator's activity channels perception data processing, and handling (output) capabilities. The total channel capacity can be grouped into the following activity channels:

(1) Communication
Direct verbal, via radio telephony (RT) or other means

(2) Oberservation
Encompassing the perceptive capabilities

(3) Information processing
Decision making and other complex or higher order capabilities

(4) Manual output activities

By design of a part-automated system it could be achieved to distribute the operator's activities over his channels in that way that the advantage of each channel could be utilized, resulting in a reduction of total time consumtion.

Our investigations in ATC meanwhile led to the specification of a part-automated ATC system rendering an increase in system capacity of 30 % compared with a non-automated ATC system.

For task fulfilment of the human operator in man-machine systems there exist two determinating and critical factors:
(1) the simultaneous information processing capability, and
(2) the time dependent information processing capability.
Although both factors cannot be seen as being independent from each other, they are described separately herein.

2.1 The simultaneous information processing capability

There is a limit concerning the number of informations - random information - man can depict and process simultaneously. This limit is given by the wellknown "magic number 7", or 7 to 8 informations. This has been measured repeatedly, and has been reported by Hermann /1/.

In the ATC system it was measured as the number of aircraft a radar controller was able to control simultaneously. Reason: with the present ATC system (in use up to 1975) the radar controller has to memorize which radar target belongs to which aircraft (call sign); what is particularly loading, because of the continuously changing constellation of the targets' positions. In this situation the human operator's short term memory for perception patterns is limiting the simultaneous channel capacity (control capacity) to 7 to 8 aircraft.

This limit can be overcome - and is done so in part-automated systems - by automatically attaching digital labels to the radar returns and therewith identifying them as determined aircraft /2/.

Generally the human operator's limit for simultaneous information processing depends upon pattern perception, attention capacity span and short term memory.

This limitation can turn out in fighter design to be the decisive criterion to select a two seat cockpit. This limited information processing capability of man can become critical, if for instance an attack mission phase for a single seat CAS fighter A/C is designed, imposing simultaneous high activity workload on the pilot in: A/C handling and steering plus weapon delivery systems management plus target acquisition.

In the attack phase there is a high probability that an overload in simultaneous information processing capability occurs concerning visual perception and manual activities.

Our design goal must be to specify the equipment handling and mission function requirement, that it not exceeds this capacity limit of the pilot. If this is not possible, a two men crew will be required.

2.2 The time dependent information processing capability

The second factor, the time dependent information processing capability, can also become a limiting factor. This occurs in a situation when the human operator is unable further to increase his working capacity per time interval without making errors. In recent studies /3,4/ this limitation is coming up in conjunction with aviation systems, and was expressed by the term "time saturation".

In a number of investigations concerning the work of the air traffic controller in this operational environment /5/ we found how it happens that the human operator approaches time saturation in a given time interval. Figure 2 shows the increase of the load within the channel groups "RT" communication" and the joint channels' observation, information processing and manual activities, related to the number of aircraft under control.

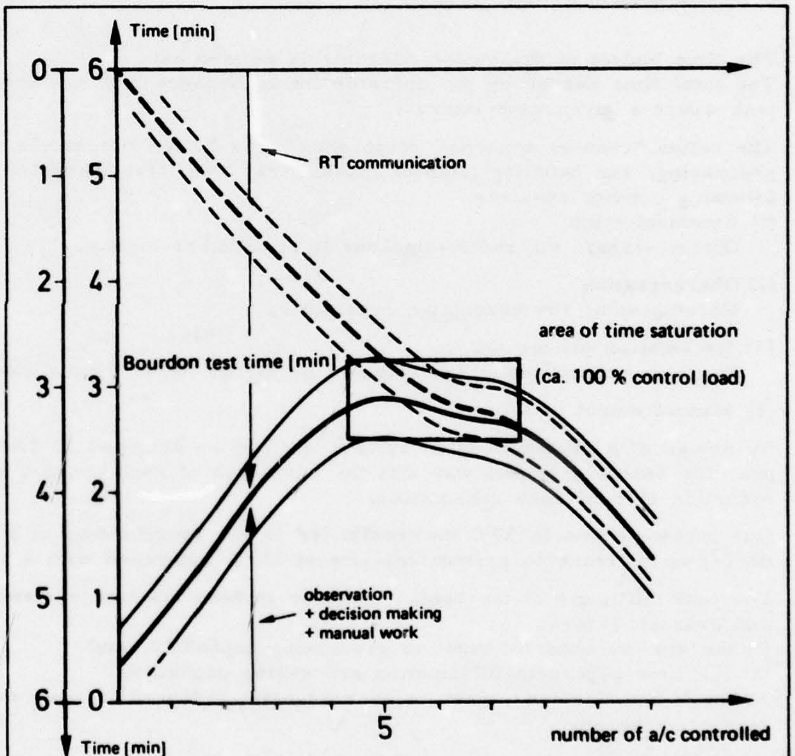

Figure 2: Example of a measured time saturation condition/Approach control | MBB

Note: "Bourdon test time" in figure 2 refers to a secondary task applied during the measurements to directly measure the "free capacity" depending on the workload. This was done to distinguish between "free capacity" and "information processing" time.

From figure 2 it can be seen very clearly:
When the number of tasks (functions) to be performed per time interval exceeds the total channel capacity of the human operator, the system becomes overloaded, and thereby unsafe.

2.3 The profit of specifying the system in accordance with the operators time-budget

Table 3 shows the gain in system capacity possible by automation, if the appropriate means are applied to really make optimum use of man's capabilities in the four channel groups defined.

The 1975 data in table 3 show the time budget per aircraft as related to the system capacity. 88 seconds workload per aircraft allow a traffic flow of 41 aircraft per hour.

The 1982 data show the design goal for the ATC system specified for 80's. It allows a traffic flow of 53 aircraft per hour, equal to 30 % gain in system capacity. The operational equipment requirements account for this time-budget in terms of:

- task assignment to man and machine specified
- information display requirements specified
- man/computer interaction requirements specified.

The same approach as chosen for evaluation of existing ATC systems, and for specifying future ATC systems can be applied to other man-machine systems; so it also applies to A/C systems. The appropriate measures for designing an aircraft cockpit can be directly developed and estimated from the measurements on ATC systems, showing the human operator's characteristics of workload (time consumption) in the four channel groups, imposed on the operator within the time interval.

DIMENSION	1975 in sec.	1982 in sec.
R/T communication	44.1	24.6
Observation	13.2	18.4
Decision making	24.7	17.7
Manual	6.2	7.5
per mean A/C:	$\sum = 88.2$	$\sum = 68.2$

Table 3: Radar Controller time-budget per A/C MBB

2.4 The time-budget approach in aircraft system design

In a two seat fighter A/C development, the time-budget of the pilot and naviator has been derived on the basis of a mission analysis program. The channel groups could be isolated similar to those, used also in the ATC work analysis:

1) Communication
pilot: internal and ATC communication
navigator: internal and guidance and control communication.

2) Manual activities:
system setting, mode switching, data input sequences

3) Visual activities (observation):
internal and external scanning, instrument reading, search for target detection.

4) Control activities:
target acquisition, updating, navigation and target acquisition error assessment, etc.

5) "Free capacity"
Time, in which the operator is not loaded in the above channel groups.

Table 4 shows the time-budget of the pilot and navigator in a straight pass auto attack (200 ft height, 600 kts) with 25 nm distance, and 150 seconds time to go. The total channel activity time does not exceed 100 percent, because the mission phases evaluated concern autopilot steered modes, in which less parallel activities are registered.

The time budget of the navigator shows 80 % load, because he is performing the target acquisition update at first via radar and secondly via LLLTV. The pilot's time budget is only loaded 50 %, because he is performing target search visually only when reaching an appropriate distance (9 nm) to target. The target acquisition update is performed by the pilot via head-up display.

Nr.	Channel group	Straight pass auto attack PILOT activity time in %	NAVIGATOR activity time in %
1.	Communication	3.3	6.7
2.	Manual activities	10.0	14.7
3.	Visual activities (extern./intern. scanning)	22.0	43.3
4.	Control activities	14.7	16.0
5.	Free capacity	50.0	19.3
	time available:	100.0 %	100.0 %

Table 4: Time-budget of the pilot and navigator in a straight pass auto attack (150 sec) MBB

For this particular mission phase a time budget for a single seat cockpit has been estimated on the conditions, that the pilot performs a straight pass auto attack with:

1) target detection via radar, with one radar update
2) Visual target detection, with one update via HUD.

The comparison of the pilot's time budget between the two seat version (from table 4) and the single version is shown in table 5.

It can be seen, that the load in the channel group (2) to (4) increases considerably. In this particular case, the pilot was loaded with target acquisition via radar plus acquisition via HUD, performing one update in each of the acquisition phases.

It can also be seen from this comparison, that the greatest increase is given concerning the manual activities. We know meanwhile, - from other systems - that overload of the operator's time budget is most likely to occur first in the manual activities channel. Automation easily leads to an unacceptable amount of "button pushing", being required for man/computer interaction.

Nr.	Channel group	Pilot activity time in %	
		single seat	two seat
1	Communication	4.7	3.3
2	Manual activities	21.3	10.0
3	Visual activities	33.3	22.0
4	Control activities	22.0	14.7
5	Free capacity	18.7	50.0
	time available	100.0	100.0

Table 5: Comparison of the pilot's time budget in a two seat versus a single seat cockpit MBB

Therefore, in view of the time budget considerations, our main concern for the cockpit design must be:

1) Limit the routine work to be performed by the aircrew, by automating routine functions.
2) Limit the information perception requirement for the aircrew
   - by presenting information of higher order (the order needed for decision making), e.g. pre-processed instead of raw data, and
   - by presenting the information required on a limited number of displays, by using "integrated equipment" (one display for a number of systems).
3) Limit the manual output required for the aircrew by:
   - providing controls, by which functions of higher order can be launched, e.g. start a given fix-updating procedure by one pushbutton, instead of by setting a number of individual equipments.
   - providing integrated control equipment, such as multi-function keys with semi-automated address information incorporated into the display format.

3. Requirements for the future cockpit design

A tactical requirement for the next fighter A/C generation does not yet exist. But even for experimental studies to determine the technological design basis, we need to assume tactical requirements, and the resulting A/C performance requirements. The tactical and performance scenario determines:

- the system/equipment requirements
- the information processing requirements
- the environmental conditions

Performance requirements assumed:

- all-weather capability
- low level CAS
- high precision in airborne navigation
- extreme manoeuvrability using control modes as direct aiming mode, lateral displacement control mode, direct lift mode.

3.1 Cockpit design technology consequences

1) The manoeuvrability requirement must be seen in conjunction with high, and with unconventional g-operations. The consequences are:
   - variable tilt ejection seat
   - limited front panel space
   - limited field of view during high g-operations
   - trade-off concerning side stick vs. center stick control, and different transfer functions (force/displacement control).

The further consequences are not discussed in detail here. To illustrate the panel space problem quite clearly, the following example is given:

The single seat cockpit for a fighter aircraft with high manoeuvrability and g-load has about 24 % of the panel space given in a two seat aircraft.

In our present two seat fighter aircraft the fuel system indicators and controls occupy 2.3 % of the total panel space available.

In the future single seat fighter aircraft these indicators and controls would use up 9.6 % of the total panel space available.

This is unacceptable!
2.3 % of the single seat cockpit space (as given in the two seat version) would require the fuel system indicators and controls to be reduced to one fourth of their present size.
This shows the relationship.

2) Man's information processing capability is a given capacity (see paras 2.1 and 2.2). The consequence is to make better use of man's information processing capability by applying means described in the following.

3.2 The technologies applied

3.2.1 The integrated versus isolated systems/equipment concept

The isolated cockpit equipment concept is determined by:

- dominance of discrete display and control equipment for each A/C sub-system
- limited centralized data processing (e.g. air data computer), and separated from that, more decentralized data processing (e.g. terrain following radar)
- consequently preprocessed information is not available to the extent desired.

The integrated cockpit equipment concept is determined by:

- complex data management with a mixture of centralized (main computer) and, decentralized (mini-computer) data processing
- use of combined displays and control panels as determined according to the time-budget and to simultaneous information processing capabilities of the pilot, respectively the aircrew.

In present day cockpits, we already have a certain amount of integrated equipment, e.g. the "weapon aiming mode selector panel", for various weapon aiming and delivery modes. The "combined radar projected map display", and the "TV/Tabular Display" with it's "Multi-Function-Keyboard", and multifunction display. But this is only a first step towards the "integrated cockpit".

Table 6 shows a comparison of the isolated vs. integrated systems/equipment concept.

From table 6 it can be seen, that the critical criterion of the integrated cockpit systems/equipment concept is the fact, that we need to provide a considerable number of "multi-function keys". This constitutes a reliability risk for manual activities, and therefore requires special attention in the phase of lay-out, and design specification.

3.2.2 Mission/flight phase dependent display of information

A second step to the integrated cockpit is to provide a certain amount of information, and to provide a certain number of controls on a timely basis. We must identify:
- mission phases, and
- flight phases.

requiring a given amount of information, defined by the individual systems and their respective functions involved.

The mission/flight-phase dependent information/control package shall be selectable by "phase-select" keys or "selectors" on a multi-function keyboard, like those on the WAMS (weapon aiming mode selector) or on the TV/Tabular displays.

Flight phases identifiable are:

- take-off and initial climb
- cruise
- terrain following
- initial, intermediate, final approach and landing, etc.

Mission phases identifiable are:

- navigation updating
- dash-in, loitering, etc.
- air-to-ground attack, selectable according to type of attack
- air-to-air attack, selectable
- weapon management in preparation for an unplanned attack
- "weapon package" selection in conjunction with a pre-planned attack, etc.

It is not considered herewith to limit the pilot's "degrees of freedom" in decision making for his conduction of flight, or mission planning and processing.

Rather more this is a definite limit to the degree of automation acceptable to be designed into the cockpit:

Leave the decision functions to the pilot (aircrew)!

Criteria	Cockpit concept: isolated	assessment	integrated
	qualitative	quantitative	qualitative
panel space required	–	>	+
scanning area	–	>	+
manual control activities area	–	>	+
number of individual displays required	–	>	+
number of individual control panels requd.	–	>	+
growth capacity for display of information	–	<	+
information availability and access required	○	=	○
display of preprocessed information	limited or –	<	great or +
function of individual controls	single or +	<	multiple or –
number of manual actions required per operational phase	–	>	+
use with variable seat tilt (field of view)	limited or –		good or +
	1 + 1 ○ 9 –		9 + 1 ○ 1 –

Table 6: Assessment of isolated vs. integrated cockpit systems/equipment concept

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This requirement has sometimes been neglected. If it is neglected, the consequences would be only a shift of workload for the pilot. The consequences were already described by Hopkin /7/.

3.3 Determination of display/control modalities for application in the integrated cockpit

The integrated cockpit is not designed by adding equipment to equipment, and arranging the items in a sophisticated way on the panel space available.

The integrated cockpit requires identification of all individual display/control requirements per individual aircraft and avionic system, and their allocation to the display/control modalities as specified in figures 3 and 4.

The modalities for display of information can be explained as follows:

1) Permanently available information encompasses information continuously needed according to the decision criteria:
   - used continuously for flight control
   - used continuously for thrust control
   - used continuously for navigational control.

2) Manual selection of information
   This encompasses all information, not already convered by (1). All information, which is not permanently displayed must be available "on request" via selection by the pilot.

3) Automatic display of information according to phase selected.
   This provides the capability to select an information/control package per mission or flight-phase. The "phase-selectors" are continuously available function keys.

4) Forced override.
   This modality ensures immediate display of information required in conjunction with:
   - flight safety (central warning)
   - enemy threat (ECM)
   - mission guidance (data link)

This concept requires a new concept for the central warning system as well as for ECM. The ECM concept is treated within the paper of Eckl /9/ (following this presentation).

Mission guidance information via tactical data link is another type of information required for immediate display.

A similar concept was already specified in conjunction with the future ATC-system /8/. By this concept optimum use can be made of display space available, without interfering with other information displayed. The modalities for manual control functions follow the same concept. The scope of the paper precludes a further detailed discussion.

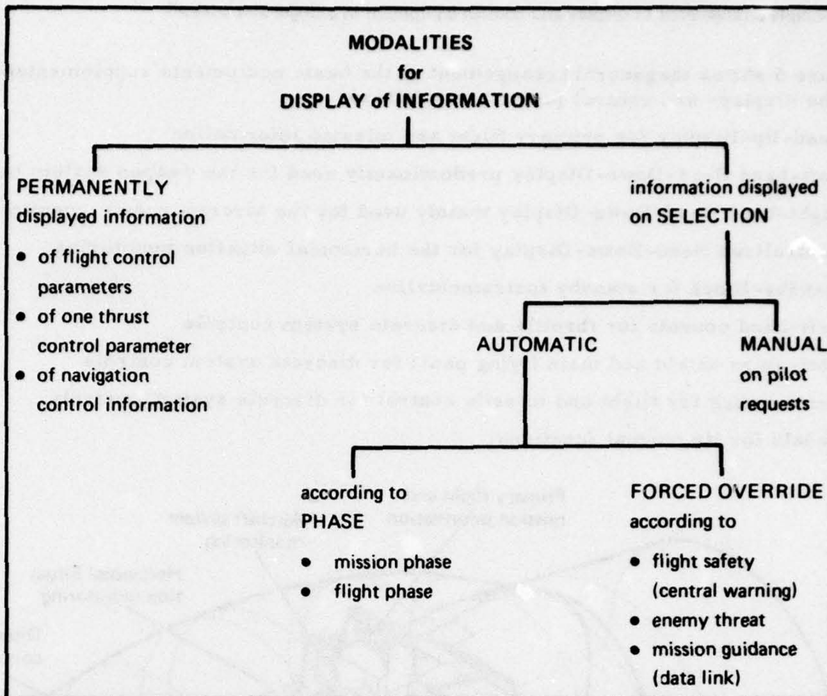

Figure 3: Modalities for display of information

The integrated cockpit concept requires a limited number of conventional and electronic displays as well as a limited number of conventional and multifunction control panels.

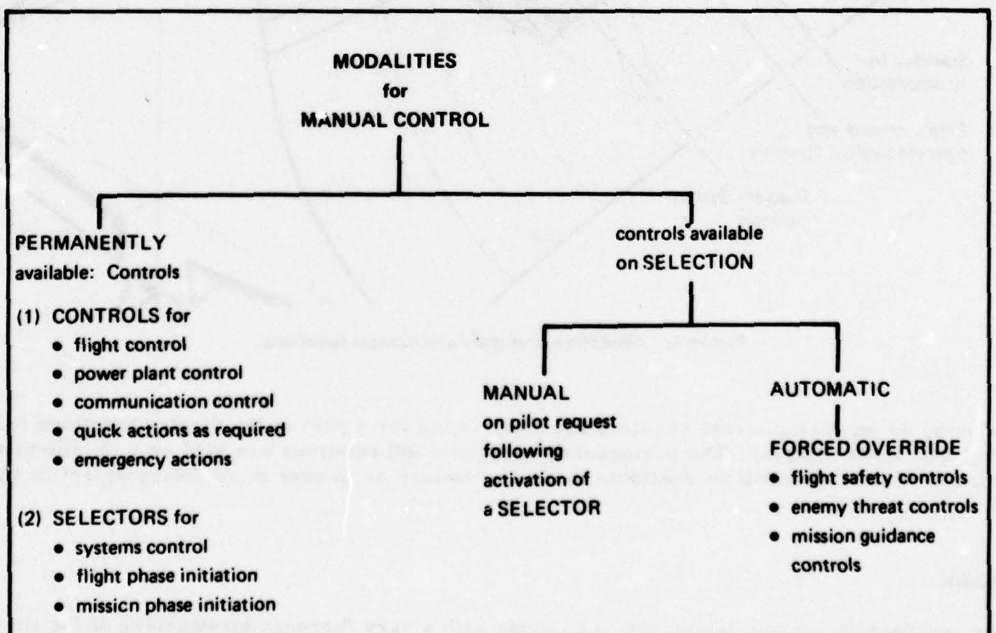

Figure 4: Modalities for manual control

3.4 Principle arrangement of display and control equipment in a single seat cockpit

Figure 5 shows the general arrangement of the basic equipments supplemented by a coarse allocation of the display- and control functions as follows:

- Head-Up-Display for primary flight and mission information
- Left-hand Head-Down-Display predominantly used for the weapon system management
- Right-hand Head-Down-Display mainly used for the aircraft system monitoring
- Centralized Head-Down-Display for the horizontal situation monitoring
- Standby-Block for standby instrumentation
- Left-hand console for throttle and discrete system controls
- Anti-glare shield and main flying panel for discrete system controls
- Center stick for flight and missile control for discrete system controls
- Pedals for its normal functions

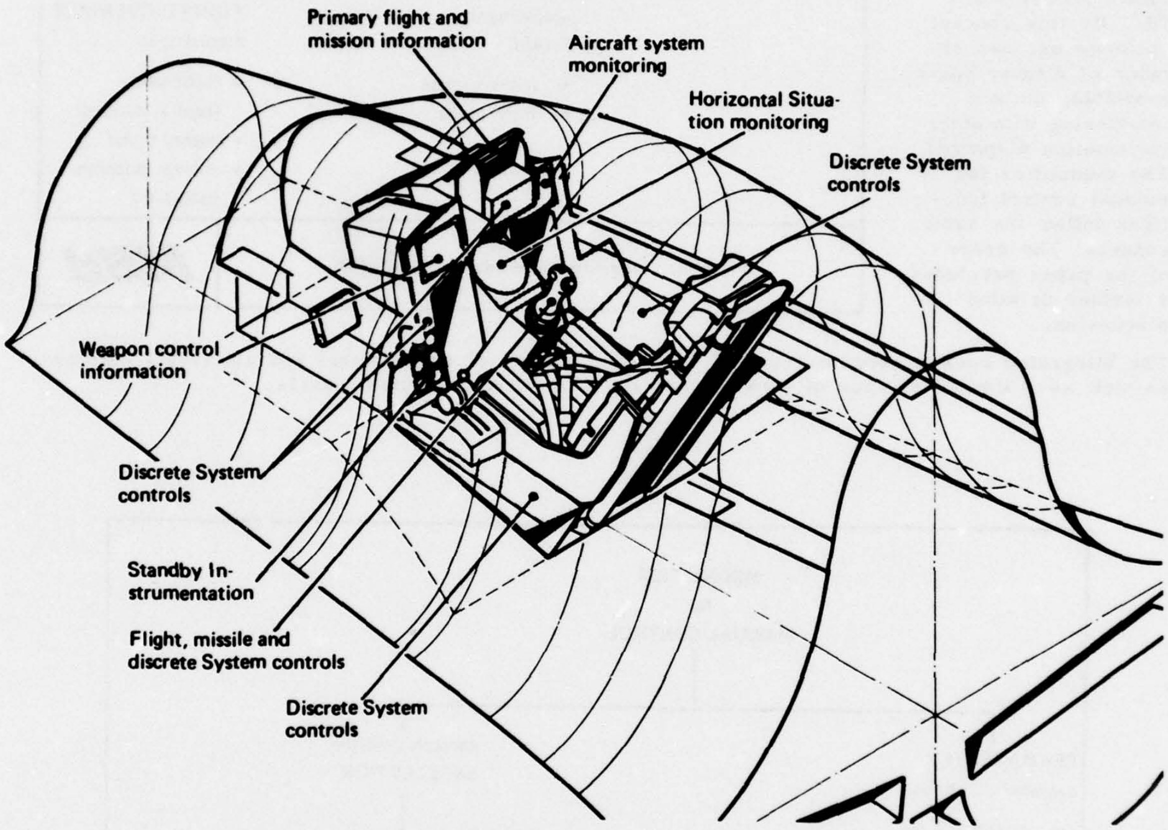

Figure 5: Allocation of display and control functions

Table 7 shows, as an example, the required characteristics for a part of the display functions for the right hand Head-Down-Display. The permanently displayed informations are held on a minimum although detailed information will be available in case of failure or in case of voluntary selection by the pilot.

3.5 Final remark

The future approach in cockpit design has to go along with a very thorough bookkeeping of the time budget which takes into account each concept variant and each change during development.

Equipment	Functions	Modalities: Permanently	Modalities: On Selection – Automatic – Phase	Modalities: On Selection – Automatic – Forced O'ride	Modalities: On Selection – Manual	Information READY for decision. Which?	System READY for action. Which?
3. Right-hand Head-Down-Display	3.1 Engine system						
	3.1.1 Speed of rotation	X					
	3.1.2 Diagnostic display as such				X		
	3.1.3 Diagnostic display in case of failure			X		Types of limitations, applicable measures	
	3.2 Hydraulic system						
	3.2.1 System pressure	X					
	3.2.2 Diagnostic display as such				X		
	3.2.3 Diagnostic display in case of failure			X		Types of limitations, applicable measures	
	3.3 Fuel system						
	3.3.1 Fuel quantity	X					
	3.3.2 Fuel flow	X					
	3.3.3 Fuel consumption calculations				X	Calculations for fuel consumption for various conditions	
	3.3.4 Fuel feeding system display as such				X		
	3.3.5 Diagnostic display in case of failure			X		Types of limitations, applicable measures	Change of feeding sequence
	etc.						

Table 7: Example for the required characteristics of functions for the right-hand HDD

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APPLICATION TECHNIQUES FOR DIGITAL FLIGHT CONTROL SYSTEMS

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SUMMARY

This publication discusses some system design and implementation considerations in application of digital processing and signal transmission techniques to flight control systems. It covers the digital flight control design cycle and both the overall similarity with analog implementation and the differences associated with software requirements specification. This is a brief overview of some of the design problems that result from the digital processing. It discusses software development by first defining software and then by showing how the software design is specified and implemented. It also describes testing at both the software and system development levels.

First the article briefly describes the three-channel redundant flight control computer used on the Boeing YC-14 advanced medium STOL transport (AMST) and provides examples of the advantages of digital applications, along with specific design solutions and software development procedures

INTRODUCTION

Use of digital flight controls has passed the experimental stage, and its application should expand rapidly in the future. Development programs such as the USAF digital multimode flight control system on the A-7 (ref. 1), the NASA triplex fly-by-wire system on the F-8 (ref. 2), and the triplex 737 system (ref. 3) used in the NASA terminal-configured vehicle program have demonstrated the application of this technology. Production systems such as the digital flight control and autothrottle on the Swedish JA-37 (ref. 4) and the PANAVIA MRCA autopilot and flight director (ref. 5) have proved that the state-of-art systems are ready for application. Completion of a successful flight test program with the YC-14 advanced medium STOL transport (AMST) aircraft has demonstrated the viability of using a three-channel digital flight control system for flight-critical control functions.

As the functional requirements of the flight control system increase, and as the mission reliability and safety requirements dictate the need for redundant systems, digital mechanization of the flight control electronics is preferred. Advantages of the digital mechanization are:

(a) Digital signal transmission and multiplexing and the ability of the digital computer to time share hardware elements reduce the quantity of hardware required for management of fault tolerant systems.

(b) The precision of digital processing, combined with the ability to employ signal selection on all input sensors, allows identical tracking of multiple channels. Large tolerance buildups, which complicate the control and monitoring of redundant servos, are avoided.

(c) With the digital computer's superior capability to execute logic functions, more complete and less costly system test is possible. In flight, the built-in test and monitoring functions permit easier detection of failures so that they can be isolated and identified for postflight maintenance.

(d) Flying qualities can be improved by using control laws not feasible with analog systems because of processing and accuracy requirements.

(e) There is improved flexibility to make changes during both the development and service life of the system.

The advantages of digital flight control systems, when compared with analog systems, have been attributed largely to the stored programs' increased computational accuracy and flexibility. However, the software associated with this flexibility represents a "new" element in the flight control system design process. New management techniques, requirements specification, and development and test methods ensure orderly, timely, error-free, and economical development of the software.

YC-14 DIGITAL FLIGHT CONTROL SYSTEM

This brief description of the YC-14's digital flight control system (ref. 6) provides background for the discussion on application techniques which follows. The Boeing YC-14 is a prototype tactical transport designed to demonstrate the feasibility of combing short takeoff and landing (STOL) performance with high speed, high altitude cruise performance. Figure 1 shows the electrical flight control system schematic. The system uses three identical general-purpose digital computers and appropriate redundant sensors, servos, and power supplies to ensure fail-operational/fail-safe operation. Each computer has the following characteristics:

Clock rate	1.4 mHz
Frame synchronization	Every 10.0 ms
Typical instruction time	4.0 µs
Inputs per channel:	
Direct current (dc) analog	8
Alternating current (ac) analog	34
Discretes	70
Serial digital	2
Outputs per channel:	
Direct current (dc) analog	15
Discretes	100

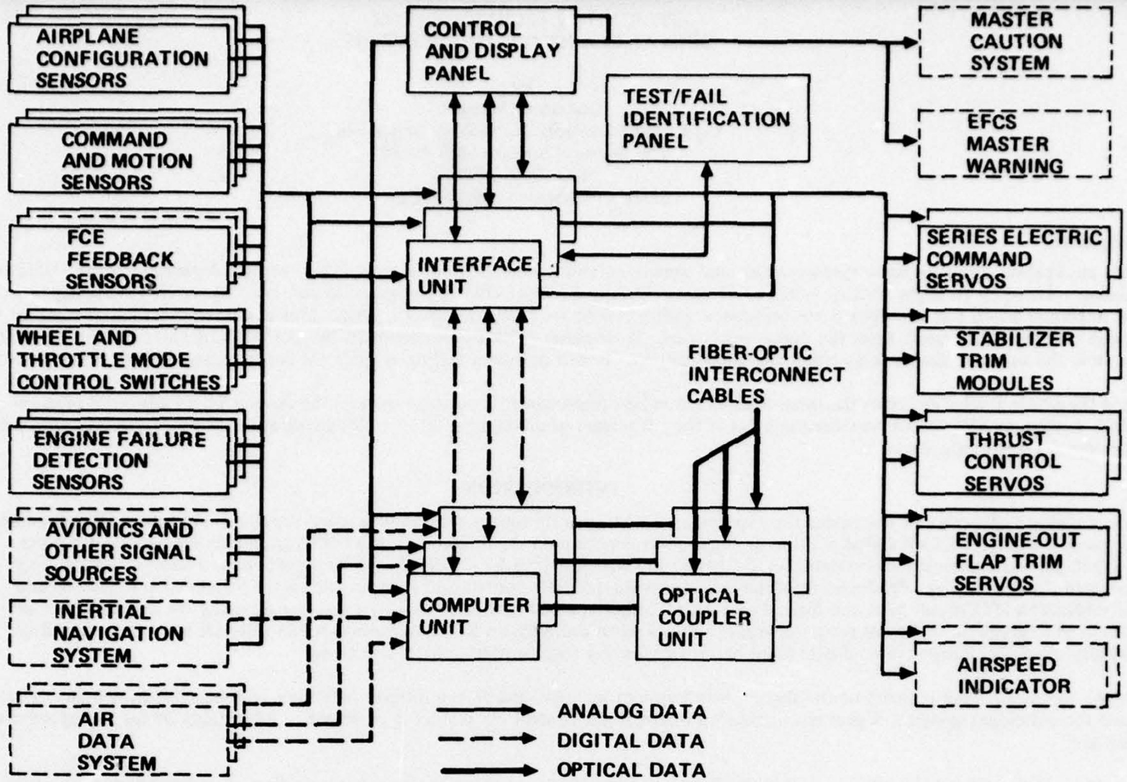


Figure 1. YC-14 Electrical Flight Control System Schematic

Inputs to the computers come via parallel digital transmission lines from interface units and via serial digital transmission lines directly from one inertial navigation system (INS) and three air data systems. Analog and discrete signals from aircraft-motion, control-feedback, pilot-control, and aircraft-configuration sensors are converted to the proper format in the interface units.

The computer transmits outputs as parallel digital data to the interface units, where the appropriate signal conversions are accomplished. Servo loops are closed in analog form through the interface unit. An exception is the throttle servo, in which a digital encoder provides position feedback, and loop closure is accomplished in software.

All cross-channel data required for redundant operation of the digital system are transmitted between the computer units via optical data links. Each sensor and switch input signal is transmitted cross-channel for signal selection such that an identical signal is selected for processing in each computer. Each signal selector is implemented in software and has an associated monitor for failure detection, fault isolation, and annunciation.

A control and display panel (CDP) and a test failure identification panel (TFIP) are the crew's interface with the system. The CDP has system-start, channel engage/disengage, and flight-mode switches, as well as status displays. The TFIP is used to run automated preflight tests of the system and provides a display for identification of failures detected in flight.

The flight control electronics hardware and most of the first issue of software were developed by Marconi-Elliott Avionics, Ltd., of England. The functions provided by the system are:

(a) Command and Stability Augmentation
The system augments the mechanical control system by addition of pitch-and-roll control wheel steering.

(b) Speed Hold and Flight Path Control
For STOL operation, the engine thrust and the deflection of the upper surface blown (USB) flap are modulated by the electrical flight control system (EFCS), thereby allowing the pilot to fly the airplane using normal piloting techniques. "Backside" techniques, usually required when operating at less than the minimum drag speed, are not used. This not only cuts out the need for special piloting skills, but also reduces the crew workload significantly and improves touchdown dispersion.

(c) Automatic Pitch-and-Roll Trim

(d) Automatic Engine-Out Control
If an engine fails at STOL speeds, the USB flaps, main flaps, and direct-lift spoilers are automatically reconfigured to establish maximum climb performance configuration.

(e) Attitude Trim Mode
For the low-altitude parachute extraction system (LAPES) delivery of cargo, the aircraft must fly 5 ft above the ground with a level cargo deck. The EFCS provides an attitude trim system which allows flight in this deck attitude at the required delivery speed and at the existing aircraft weight. The combination of speed hold, a tight pitch attitude control, and automatic pitch trim minimizes the transients as the cargo leaves the aircraft, and simplifies a normally demanding task.

(f) Altitude-Hold and Heading-Hold Pilot Assist Modes

(g) Refuel Mode
A special attitude demand system is provided for aerial refueling, which is designed to ease small altitude adjustments when flying close to the tanker aircraft.

(h) Flight Test Programmer Modes
This function allows selection of modified control laws for flight test evaluation. With separate memory allocation, reprogramming has minimum impact on the basic software.

(i) Preflight Test
Automatic and semiautomatic modes are provided for test of the flight control electronics and the interfacing sensors and servos.

Figure 2 shows use of flight computer memory to accomplish these functions.

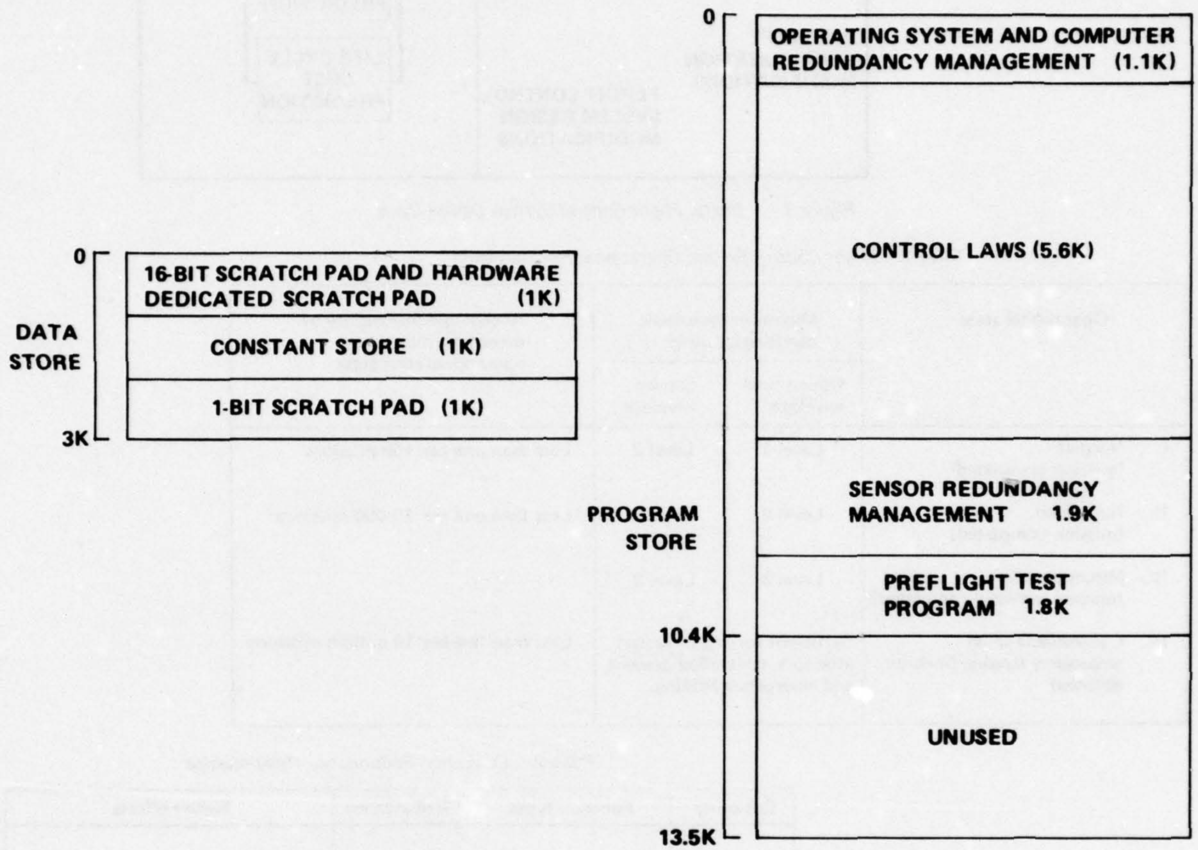


Figure 2. YC-14 Flight Control Computer Software Memory Allocation

A flight test instrumentation bus in channel 2 makes all cross-channel data and additional analog data available for magnetic tape storage during flight. This provides digital quality data for these signals, required for analysis of the system behavior.

FLIGHT CONTROL SYSTEM DESIGN REQUIREMENTS

General

Figure 3 shows the flight control system design cycle. The procuring agency defines the aircraft missions, their combinations, and success criteria. Detailed specifications for functional readiness and system maintainability are also provided. Operational requirements are expressed in terms of handling quality levels under normal and failure conditions, as summarized in table 1.

The criticality of the functions, in terms of their failure effects and the ability to complete the mission following loss of the function, is a basic factor determining the level of system redundancy. Three levels of criticality are defined: (1) The function is critical if loss of the function results in an unsafe condition. (2) It is flight-phase critical if loss of a function results in an unsafe condition during specific flight phases. (3) It is noncritical if loss of the function does not affect flight safety. Criticality-redundancy relationships based on state-of-the-art reliability numbers are illustrated in table 2.

Studies such as those described in references 7 and 8 have analyzed the design of fault-tolerant redundant flight control systems. The concepts developed allow use of in-line monitoring and system reconfiguration techniques to enhance the probability of detecting, isolating, and recovering from failures. The results modify the criticality redundancy relationships, shown in figure 2, by allowing many of those failures listed as fail-safe to become fail-operational. Depending on the mission and the system failure requirements, this may allow the use of a lower level of redundancy than shown.

For the YC-14, there are no critical functions; however, early simulator studies indicated that loss of augmentation during STOL landing degraded the aircraft handling qualities to a level unacceptable for normal operation in adverse weather. Augmentation for the STOL regime was treated, therefore, as a flight-phase-critical function. Only a triplex system could provide the required functional reliability and failure-mode protection.

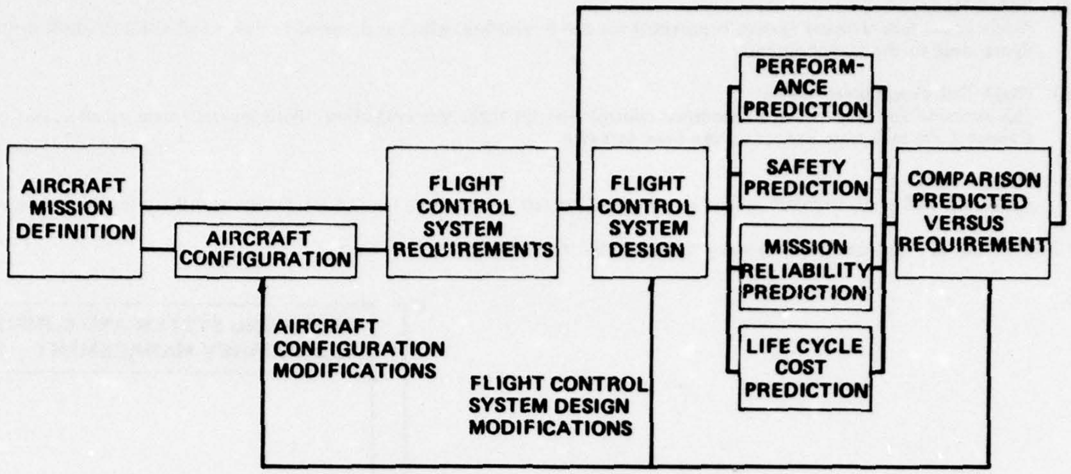


Figure 3. Digital Flight Control System Design Cycle

Table 1. Flight Control System Operational Requirements

Operational state	Minimum acceptable handling qualities		Acceptable probability of degradation to lower operational start-state
	Operational envelope	Service envelope	
I. Normal (mission completed)	Level 1	Level 2	Less than one per 100 missions
II. Restricted (mission completed)	Level 2	Level 3	Less than one per 10,000 missions
III. Minimum safe (mission partially completed)	Level 3	Level 3	
IV. Controllable to an emergency landing (mission aborted)	Sufficient for engine restart attempts, controlled descent, and emergency landing.		Less than five per 10 million missions

Table 2. Criticality/Redundancy Relationships

Criticality	Function types	Redundancy	Failure effects
Critical	Full-time augmentation Flutter suppression Fly-by-wire	Quadruplex	First fail—fail operational Second fail—fail operational Third fail—fail safe
Flight phase critical	Augmentation Maneuver load alleviate Category III autoland	Triplex	First fail—fail operational Second fail—fail safe
Noncritical	Category III autoland Cruise autopilot	Dual or simplex	First fail—fail safe First fail—fail safe

The aircraft supplier has control over the aircraft configuration and the flight control system design needed to meet these requirements. To ensure that a minimum cost design is established, an iterative development cycle involving comparison of the predicted and required performance with a given design and design alternatives for both the aircraft configuration and the flight control system design is required. During this process, the aircraft designer identifies control system features to improve aircraft performance. For the YC-14, these features, beyond those that met the basic requirements, include:

(a) Reconfiguration of the upper surface blown flap and the main flaps in the event of an engine failure, to improve go-around performance.

(b) Addition of a control mode to allow selection of a range of desired attitudes for aerial cargo delivery without having to change airspeed.

(c) Control of the limiting deflection of the upper surface blown flap as a function of engine thrust and airspeed to minimize structural weight.

(d) Use of a speed-hold system and conventional piloting techniques to provide accurate flightpath control and minimum touchdown dispersion.

Software Requirements Specification

For the digital flight control system design and development, the basic procedures established for analog systems were followed and found to be appropriate. There was, however, one additional requirements and design process path, required for the software, parallel to and with processes similar to those established for the hardware. This parallel path rejoins the development at the software/-hardware integration and acceptance testing stage of system development, as shown in figure 4.

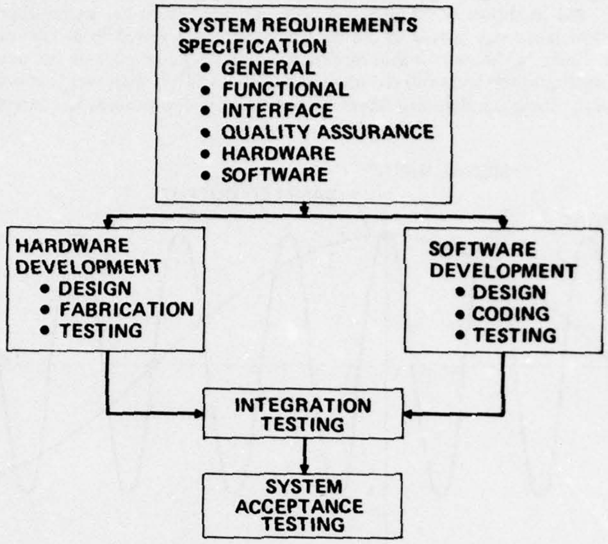


Figure 4. Digital System Design

The system requirements are the top-level specification of the system behavior. The system functional requirements define the system control modes, the control laws and mode logic, the redundancy organization, and the interface with the flight crew and supporting subsystems. These are expressed in a format which is largely independent of an analog or digital mechanization.

The software requirements are defined only to the extent necessary to ensure that performance meets the functional requirements. Various formats may be used to define software requirements: narrative description and control law block diagrams; algebraic equations, Boolean logic statements, and logic gate diagrams; flow diagrams; and sequence diagrams and data lists. While these requirements are the basis for software design, they are not coding instructions. They should be specified in a form that is easily understood by both flight control systems engineers and software engineers.

DIGITAL SYSTEM DESIGN CONSIDERATIONS

Introduction

The application of digital processors to flight controls permits the mechanization of more sophisticated control laws, logic and redundancy management system test functions, a process too complex for use of conventional analog techniques. The transition from analog to digital, however, introduces a number of unique problems to be considered during the design of the digital system. The following is a brief overview of some of these problems.

Nonredundant Design Considerations

Figure 5 shows the fundamental design considerations in implementing control law functions, using a digital system. Sensor signals will be analog (either ac or dc), digital and discrete. Signal conditioning is, therefore, required to scale and buffer these signals and place

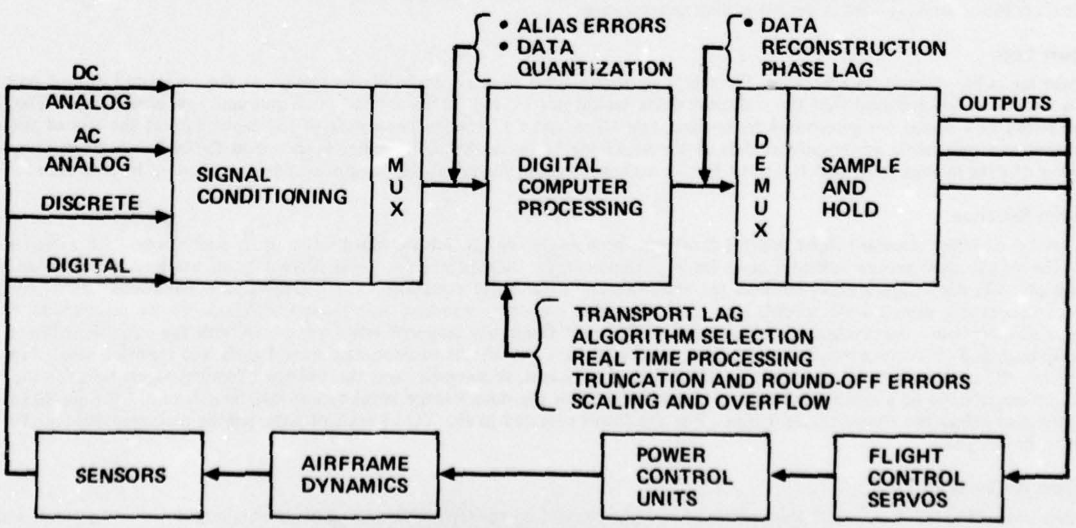
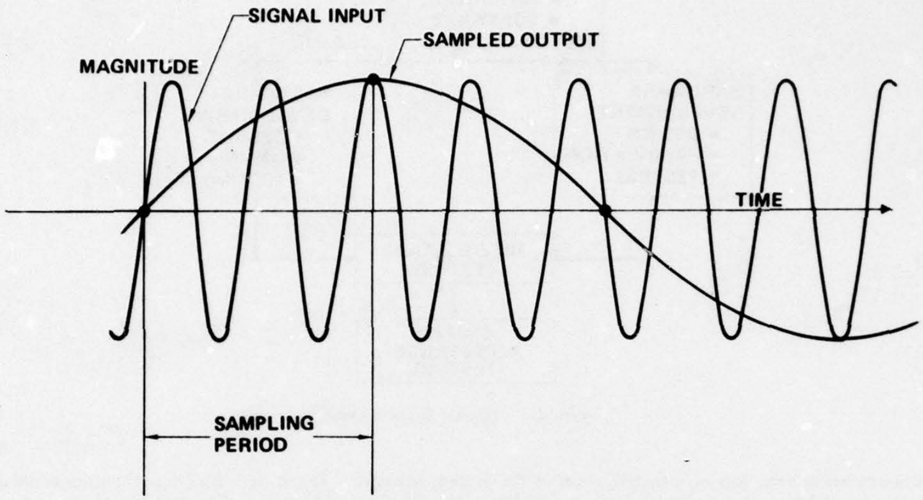


Figure 5. Digital Mechanization Design Problems

them in a digital format compatible with the particular digital processor. Multiplexing is used to cost-effectively transmit the information to the digital computer for storage and processing. Digital output commands must be demultiplexed and the analog commands constructed using sample and hold circuits. Servo feedbacks are conventionally analog and the servo drive loops are generally completed in analog form. The special design factors to consider with these processes are detailed below.

Alias Errors

Sampling of a sinusoidal analog signal at a frequency which is near a submultiple of the signal frequency can result in the generation of an apparent low-frequency signal. This is shown in figure 6. A noise component on the analog input signal can be interpreted in the same way and causes undesirable low frequency signals in the control bandpass, referred to as alias errors. These errors must be avoided by use of filtering techniques. Designing these anti-aliasing filters requires a knowledge of the noise spectrum on the input sensor signals. For the YC-14 design, all analog signals were filtered using analog input filters with two first-order lags—one with a 20 rad/s and one with a 40 rad/s cutoff frequency. These standardized filters did not result in compromise in control law design, and eliminated all aliasing effects.



Data Quantization

Because of digital representation of all input signals, there is a minimum signal difference, resulting in a quantum step of one bit of digital data. The system performance requirements dictate the range over which an analog signal is required, and the type of analog-to-digital converter used dictates the number of discrete steps used to represent these data. A further limitation may be placed on the resolution of the signal by the size of the digital data word and the scaling employed within the processor.

The 12 bit A/D converters used in the YC-14 flight control system were adequate. Since the data word used in the YC-14 flight control computers is 16 bit, the processor word size did not place additional burdens on input quantization. To allow for voltage variation and choice of input resistor at the input interface, the digital input words were scaled so that the defined input signal range represented about 90% of the digital range.

Inputs from such digital sensors as air data computers are usually defined by the sensor specification. For parameters with a large range, such as altitude where fine resolution is required, two digital words (one most significant and the other least significant) may be used. Any undesirable effects of quantization will be noticeable at the output commands generated by the processor. The effects of quantization, therefore are governed by control law gains and scaling of the output commands. The YC-14 performance requirements were specified in terms of a minimum resolution of an output command of 0.2% of the full-scale value. This endpoint requirement placed a control on the scaling, as well as on input data quantization.

Transport Lags

Transport lag is the elapsed time between the sampling of the input analog signal and the update of the computed output command. For the YC-14, it was required that the transport delay would not exceed 20 ms for the pitch roll and yaw inner loops. The 20 ms are comprised of a 10-ms computational frame time and 10 ms shared between processing of the input data at the end of the previous frame and processing of the output data at the beginning of the subsequent frame, as shown in figure 7. A more complete discussion of data management and transport lag for a three-channel, digital, flight control system is presented in reference 9.

Algorithm Selection

Computation of many standard flight control functions, such as thresholds, limits, and deadbands, is easily done with a digital computer. Use of standard proven software modules is recommended. Algorithms for linear filter implementations are available in a wide variety. Tustin's substitution method for transforming linear filter equations into the difference equations used by the digital computer has gained wide acceptance. This algorithm preserves stability, and a complex filter can be mechanized by cascading simpler ones. All methods result in some variation of frequency response when compared with the equivalent linear filter response, and all require some special attention to deal with the effects of processor word length and iteration rate. Figure 8 shows the hysteresis characteristics associated with trapezoidal, rectangular, and the bilinear (Tustin's) algorithms for the digital implementation of a second-order filter. Consideration of the data storage requirement and time required for algorithm processing also affect the choice of algorithm. For the filters required in the YC-14 control laws, simple rectangular integration proved to be adequate.

Real-Time Performance

Real-time processing is, of course, a requirement for flight control computer processing. This is obtained by use of a clock and master reset function which divides the computation time into fixed time sequences called frames. All real-time processing

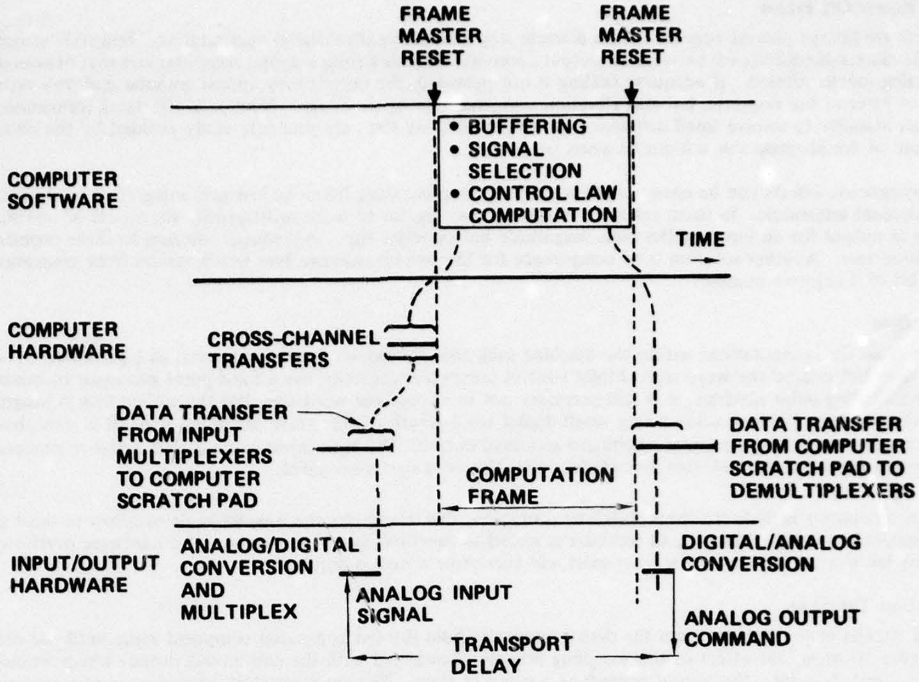


Figure 7. Input-to-Output Transport Delay

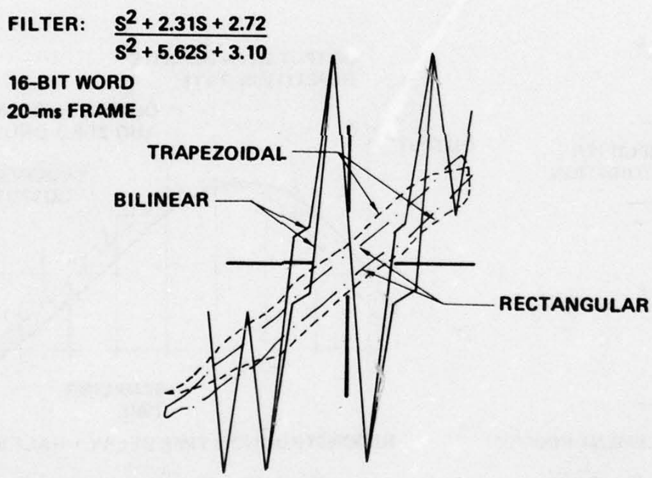


Figure 8. Filter Hysteresis Characteristics (Trapezoidal, Bilinear, and Rectangular)

must be completed within the elapsed frame time before the master reset re-initiates the computational sequence. The repetition rates at which function outputs are required to be computed are a function of the dynamic range of the system being controlled. For simple filter algorithms, a general rule is that the sampling rate needs to be 10 times the S-domain dynamic range. For the YC-14, control law iteration rates were based on use of eight frames, each of 10 ms duration as follows:

Rate	Sequencing	Function
100/sec (10 ms)	Each frame	Executive functions
50/sec (20 ms)	Every other frame	Processing of throttle encoder position feedback
25/sec (40 ms)	Every fourth frame	Pitch, roll, and yaw control laws
12.5/sec (80 ms)	Every eighth frame	Speed hold, including USB flap command control laws, gain schedules, logic functions, and failure diagnostics

Signal selection and monitoring of input signals were computed at a rate consistent with their use in the control laws. Although each logic computation was repeated only every fourth frame, the logic processing was distributed throughout the eight frames to be compatible with the associated control laws and to make the fraction of time used in each frame approximately constant. Of the total 80 ms available in the eight frames, the YC-14 used 52 ms, while the longest computation time in the worst-case frame was 8.3 ms.

Truncation and Round-Off Errors

Truncation effects are always present because of the discrete step size inherent in digital computation; however, in normal situations there is little discernible difference between an output command obtained from a digital computer and that obtained from an equivalent analog mechanization. If adequate scaling is not provided, the output may appear granular and may result in noticeable abruptness or jitter in the response, but this situation indicates inadequate design. Nonlinear effects of truncation, which can be exhibited as an inability to resolve small differences in a tight tracking task, are generally easily avoided by the choice of correct scaling and the use of double-precision arithmetic when necessary.

Some peculiar asymmetric effects can be encountered when long time constant filters or low gain integrators are being implemented with two's-complement arithmetic. In these cases, an integrator may appear to work satisfactorily for inputs of one sign, but provide zero change in output for an input of the same magnitude but opposite sign. A frequent solution to these problems is the use of a slower iteration rate. Another solution is to compensate for the one-bit negative bias which results from truncation of the least significant bit of a negative number.

Scaling and Overflow

It is necessary to scale the computations within the machine such that the value of each parameter, as represented by a data word in the machine, does not exceed the word size. Flight control computers generally use a fixed-point processor to minimize hardware cost. With a floating-point machine, it is still necessary not to exceed the word size, but the scaling task is insignificant. Figure 9 shows the overflow problem using a very small digital word length of three bits, including sign. It is seen that a ramp input signal into a gain in an analog computer results in a saturated output. The same input into a gain in a digital processor results in a maximum output of opposite sign immediately after the word sign is exceeded.

The YC-14 digital computers include hardware overflow protection that eliminates the possibility of overflow in most processes. In addition, to maximize testing efficiency, all software is scaled so overflow should not occur. If a hardware overflow protection is activated during lab tests, a software error must exist and corrective action is required.

Data Reconstruction Time Lag

Sample and hold circuits at the outputs from the digital computer hold the last computed command value until the command is recomputed. Figure 10 shows the effect of this sampling process. Compared with the continuous output which would be obtained with an infinite repetition rate, the output appears as a series of steps. The approximately equivalent continuous output obtained by joining the points representing the command values at time intervals half way through each sample is seen to give an apparent time delay equal to half the sampling time. This reconstruction time lag adds to the transport lag to reflect the overall time lag due to the digital processing.

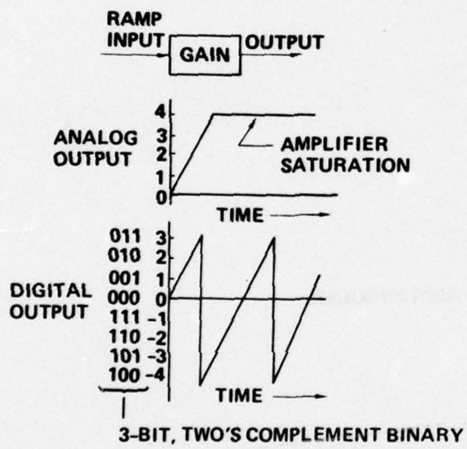


Figure 9. Illustration of Overflow Problem

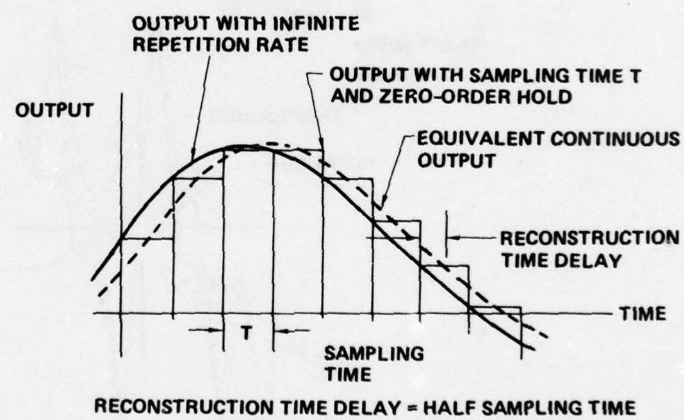


Figure 10. Data Reconstruction Time Lag

Redundant System Design Consideration

Digital system design requirements associated with redundant operation are related to computer synchronization, signal selection, failure monitoring, and data handling. The ability of the digital computer to time share hardware elements makes the cost and complexity of redundant system design acceptable. Use of identical software in each of the redundant channels can be a source for common mode failures, thus expanding the failure modes and analysis of effects. Software errors, as well as hardware failures, must be assessed in determining the system design integrity.

Computer Synchronization

Digital implementation of a redundant system involves determining whether the design will call for either synchronous or asynchronous computers. In general, designers have chosen to run the computers synchronously. The advantages of synchronous operation are as follows:

(a) Signal selection and monitoring are simplified in that tolerances do not have to allowed for data skew between channels.

(b) Command equalization to compensate for integrator drift is not required.

(c) Problems associated with disagreement in timing of mode selection changes are avoided.

(d) Voting of output commands is not required.

Synchronous operation involves the choice of synchronizing the clocks in each computer or of synchronizing the initiation of the of the computational frames. Both methods have been employed successfully. With synchronous clocks, the computers run with bit-for-bit and step-for-step identity between the channels. This allows for time-efficient voting and monitoring of inputs and out-

puts using multiplexed hardware voters. With asynchronous operation, when using software signal selection and monitoring of the input signals, the output commands also agree to the bit level, but a greater tolerance is allowed on the output timing within the frame. Both techniques require a mechanism using hardware or software, to initiate and maintain synchronism and to provide for graceful reconfiguration to a lower redundancy in the even of a clock failure or a failure in the cross-channel data transmission system. The system can be made tolerant of momentary failures of a several-frame duration.

In the YC-14 computer, the basic frame aligment is accomplished when electrical power is applied to start the computers, immediately after computer initialization. A much simplified description of the process of obtaining synchronism follows.

Figure 11 shows an example sequence of frames, nominally of 10 ms duration, in channels 1 and 2. The program reset counter, shown for channel 2, counts clock pulses equivalent to 10 ms and then issues a master reset. This master reset results in setting the computer sequence control register to zero. Comparing the frame initiation times in each channel, it is seen that channel 2 leads channel 1. A delayed software controlled reset in channel 2 allows the sequence control register to be restarted to add a frame period extension. Synchronization is established in frame (n + 1). Additional small time increments are added in frame (n + 2) to maintain synchronism. Each computer normally adds a very small increment, as shown for frame (n + 3), so that lag in the frame sequence can be corrected by omitting the addition of this increment.

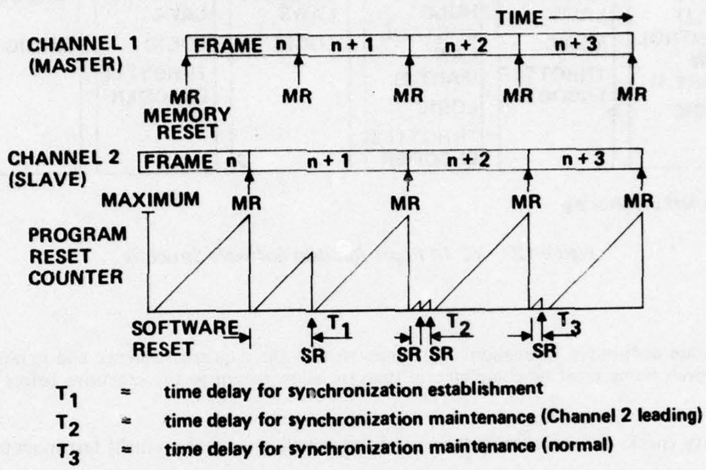


Figure 11. Frame Synchronization

A synchronization word is transmitted to the other channels with a fixed delay relative to the master reset. Times of arrival of these synchronization words are compared to establish which channel is the master and the magnitude and sign of the delays introduced into the slave channels.

Different algorithms are used for establishment and maintenance of synchronization. If two channels power up and enter a synchronization maintenance phase, the third channel can still be brought into synchronous operation. Care in the design ensures that a failing clock in one channel does not affect synchronization of the two good channels.

SOFTWARE DEVELOPMENT

The term "software," some of the different types of software, and a typical structure for flight-resident operational software are defined here prior to discussing the software development process.

Software

Software is the generic name for all of the documentation used to generate the computer memory state required to satisfy the software requirements. This includes the design specification, design and test reports, tapes, cards, files and listings. Software failures cannot occur because software, being documentation, can only have errors. Software errors can and will exist as a result of design or implementation mistakes unless a disciplined approach is followed in the software development process.

The software requirements specification precedes the development of the software, and the software requirements specification is not part of the software. It follows, therefore, that a software requirements error is not a software error. The software is error free when it satisfies all of the software requirements.

Three different types of software are:

1. **Operational Software:** This is the group of computer programs residing in the flight computer which governs the data processing associated with the flight control system functions.

2. **Support Software:** This is the group of programs employed to facilitate the development, loading, and testing of the operational software. These may include a compiler or an assembler, editors for making program changes and for linking smaller modules, a program loader, and a simulator for testing of the software in a host machine.

3. **Hardware Acceptance Test and Diagnostic Software:** This is a group of programs which, when loaded into the flight computer memory, allows verification testing of all of the digital processing, storage, and data handling facilities of the flight computer.

Figure 12 shows the structure of the YC-14 flight-resident software. The executive is the top level of the program structure. It controls the entry to the computer initialization, frame synchronization, system start, frame control, and preflight test modules. All program paths begin and return to the executive. Lateral jumps are not allowed. The frame-control module orders the program so that each frame is executed in sequence, and when the computations are completed for a given frame, the control is returned to the executive.

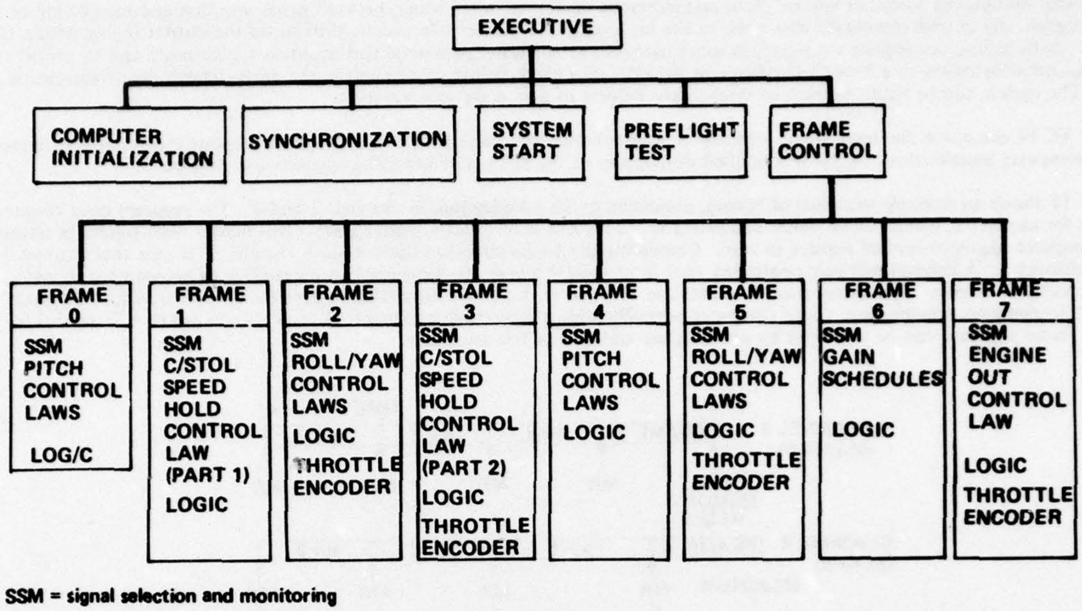


Figure 12. YC-14 Flight Resident Software Structure

The computational modules are distributed throughout the frames so that the required sequence and iteration rates are obtained. The computation within a given frame must be completed in time to allow return to the executive before the end of the allocated frame time.

The executive performs safety checks before entering the scratchpad initialization and preflight test modules.

Development Process

The end item of the software development phase is an approved, error-free set of software which satifies the software requirements specification. Development includes three basic tasks: design, implementation, and testing. The design should be completed before the implementation and testing are started. Software implementation and testing should proceed in parallel following completion of a preliminary design review.

Software Design

Software design includes those tasks required to translate the system requirements into a software design specification, such that all performance requirements are satisfied and code implementation and testing of the software can proceed. The software design specification is the detailed technical description of the operational software, written in programming terminology. It provides the definition of the software structure and the modular components that make up that structure. It also defines the standards to be used, to ensure the development of uniform software which is error free and easy to understand and to test. The software design phase includes analysis of the effects of the digitization process and the use of software design techniques to ensure that adverse effects are negligible.

Modularization of the program establishes a set of individual modules which can be linked together to form a complete program. Each module, usually less than 100 instructions, is an easily managed element which can be used for estimating both memory size and processor timing, and also for establishing scheduling and cost controls. It is documented in sufficient detail to allow programming and generation of software test plans to proceed without reference to other modules. Documentation includes an operational and design description, input and output data organizations, flow charts or equivalent to define the process, and special programming instructions.

Each module defines a function or process such that the effect of a programming change in one module will not directly affect other modules. Addition or deletion of modules can be achieved without greatly affecting the system program. Use of one of the modules for data storage, combined with use of common data terms which can be used by all modules, simplifies storage and permits modification of data without changing other modules.

Preliminary design review of the software design specifications should precede initiation of detail coding and testing.

Coding and Module Testing

Software is implemented by coding and testing on a module basis. Software testing may involve use of the flight computer, if available, or otherwise, a simulation of the flight control processor in a host computer. A test routine is developed to test every aspect of that module. If one module calls another module, the latter is simulated to the minimum extent necessary to test the link operation.

The software design specification is the basis for module implementation. Use of separate coding and testing teams minimizes the probability of a software error due to misinterpretation of the specification.

Hardware/Software Compatibility Testing (Integration Testing)

Software integration testing involves testing of the complete software program when loaded into the flight control hardware. Ground support equipment permits sufficient simulation of sensor stimuli and output servos to allow a preliminary investigation of the system operation. It permits testing of the hardware/software compatibility when the software is operating in real time with the actual hardware and an active interface.

For the YC-14, formal software control procedures were introduced when the hardware/software compatibility testing was complete.

This was expedient for the prototype program, in which the need to hold down software development costs was great. In a production program, it is expected that formal software control would be implemented following the preliminary design review.

SYSTEM-LEVEL TESTING

System-level testing verifies that the software design does indeed satisfy the system operational requirements. This testing is required both prior to first flight and prior to flights using modified versions of the software during flight test development – essentially no different from the procedures used for analog systems.

Reference 10 discusses how this was accomplished for the YC-14. Two methods of system-level testing were used:

(a) Open loop – the flight control computers are connected to a simulation of the aircraft systems with no aircraft dynamics.

(b) Closed loop – the flight control computers are connected to a simulation of the aircraft systems and the aircraft dynamics with simulated sensor inputs.

Figure 13 illustrates the semiautomatic test facility used for the open-loop tests. It enabled a large number of sensor-to-servo dynamic open-loop tests to be run to provide a high level of assurance of the correctness of the software. The procedure involved comparing the system response resulting from controlled sensor test inputs to the response predicted by an independent simulation of the hardware and software. A minicomputer, programmed in FORTRAN, was used to generate the predicted test results. A general-purpose computer and special interface equipment were used to generate the test stimuli, measure the flight computer response, and compare the actual and expected responses.

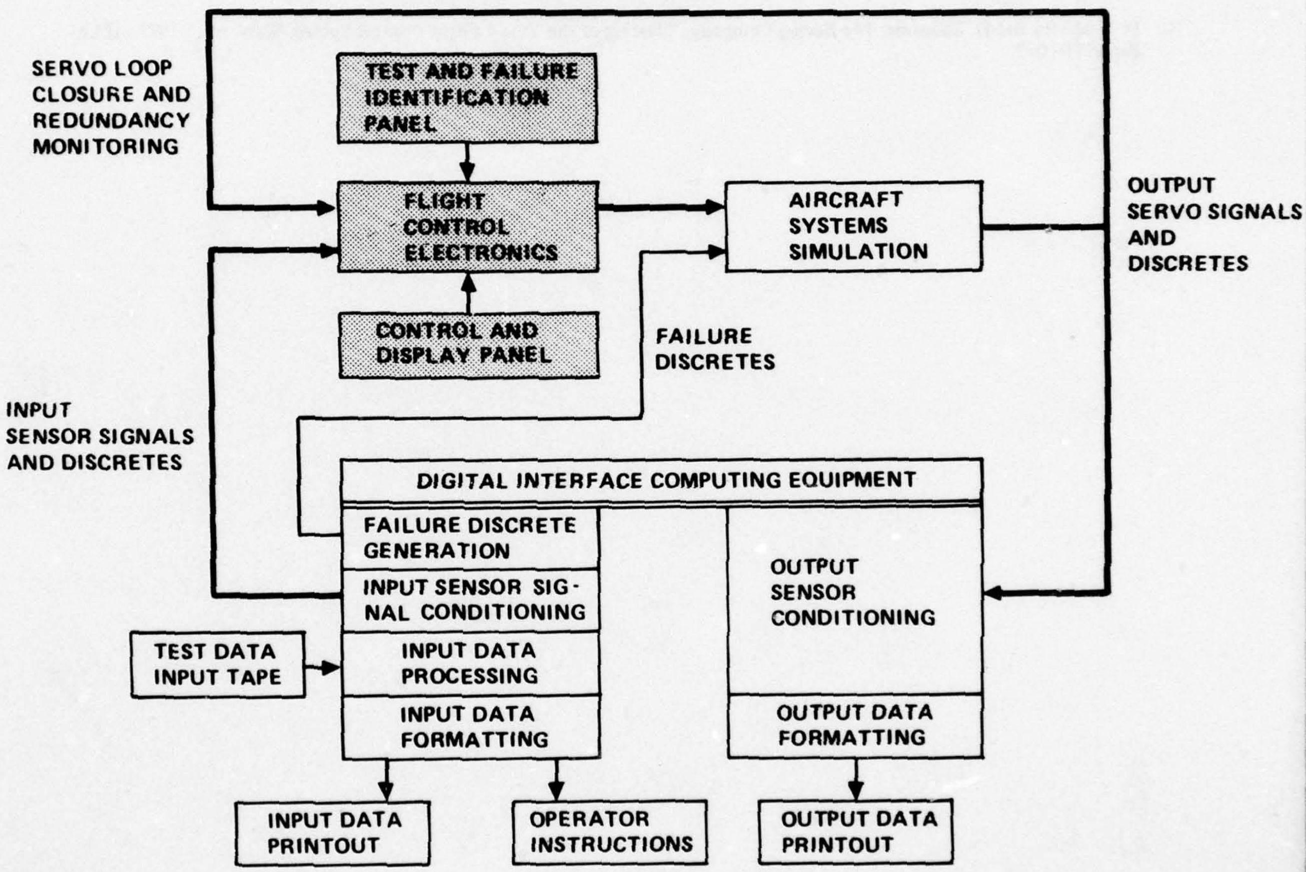


Figure 13. Open-Loop Test Configuration

CONCLUSION

The application of digital techniques to flight control provides hardware-reduction benefits in redundant systems, resulting in improved reliability and fault detection capability. The ease with which logic functions are performed by digital computers enables the use of comprehensive automated system test to reduce the aircraft maintenance burden. The flexibility to make changes enables major improvements to be introduced during flight test development.

This treatise has covered some of the special disciplines which are associated with the specification, design, and test of these digital systems, and has referenced applicable supporting YC-14 experience.

As shown, the benefits of digital flight control applications are being achieved and we predict more extensive application in the future, in both commercial and military aircraft.

Flexibility to make changes is real: however, extensive testing is required, not only to ensure that the change has been made, but to to ensure also that errors have not been introduced elsewhere in the program. Hardware or software improvements which would minimize the need for this testing would provide the benefit of the flexibility with the added dividend of low cost.

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THE DESIGN OF A HIGH 'g' COCKPIT

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Abstract

Performance improvements allow the new generation of fighter aircraft to sustain high 'g' over a large part of their flight envelope. Success in combat will be influenced by the ability of the pilot to operate under high 'g' - a new requirement which has direct implications on the design of the cockpit.

After discussing the physiological factors, the geometric aspects of reclining the pilot's seat, in order to achieve a measure of 'g' alleviation, are illustrated. The implications of such a change with respect to displays and controls are also presented. New techniques offer solutions to the problems associated with displays and controls. Radical changes in cockpit layout are implied: they require further evaluation on ground simulators and by experimental flying programmes.

1. INTRODUCTION

There is an interest in fighter aircraft design which goes well beyond that of specialists like ourselves, with a direct involvement. The majority of laymen, asked to describe a dog-fight, would turn to aviation rather than to tailwaggers. And the principles of dog-fighting, first established in World War I, refined in World War II, and re-examined in Korea, Vietnam and the Middle East, are still applicable today. Simple rules, such as having the opponent in sight at all times, still apply. Of course, better aircraft performance, and better weapons have introduced new elaborations on the old techniques, but the skill of the individual, and his ability to judge a tactical situation is still all-important.

The new generation of fighter aircraft, as we have seen this week, incorporate features which are a considerable advance on current generation aircraft. These features change the character, and the detailed tactics, of the traditional dog-fight. Earlier speakers have referred to improvements in aerodynamic design, in structural materials and methods, in avionic fit, and in weapon technology. From the pilot's point of view the most striking feature is likely to be the ability of the new designs to sustain high load factors in manoeuvres over a large part of the flight envelope.

The consequence of this manoeuvring capability is that the outcome of future fights may be determined by the ability of the pilot to operate effectively under conditions of high 'g'. A rapid deterioration of mental and manipulative skills occurs before black-out, and if this process can be delayed, a greater success rate in combat is to be expected. The outcome of dogfights is therefore highly dependent on human factor aspects which are related to high normal accelerations - visual acuity, use of limbs, and mental activity. The means of improving these aspects - posture, g suits, training, motivation all come under study, and for once, human factors becomes a central rather than a peripheral activity. The focal point of this activity is the high 'g' cockpit.

2. PHYSIOLOGICAL FACTORS

For many years, physiologists and doctors have taken pains (and pleasure) in strapping their fellow men onto platforms which move and vibrate, in order to measure their reaction to various types of motion stimuli. There is no shortage of published material concerning the influence on man of motion and vibration - translational motion, rotational motion, sitting, lying down, small amplitude, large amplitude, low frequency, high frequency. The application of these data to aircraft control problems is sometimes difficult, because of the necessay extrapolation from the laboratory environment to the airborne one, but with this in mind, the literature tells us a great deal about a pilot's tolerance to g forces.

It is well established that man in a supine position is better able to tolerate g than a man in an upright position. Reference 1 indicates that for an exposure of around 20 seconds, a pilot can withstand 4 additional 'g's comparing the vertical with the horizontal position (Figure 1). Unfortunately, the supine man is not an effective operator of an aircraft. He cannot see where he is going, and his ability to manipulate controls is greatly reduced (Figure 2) compared to the seated position (Figure 3).

As a proportion of the total flight time, the time the pilot spends at high g is small, and no compromise is allowable in ease of operation at sensibly 1'g' (for example, cruise, landing) in order to achieve better performance under 'g'. We must therefore seek a reclined position which will give some benefit - in terms of g tolerance, without any adverse effects in other flight regimes. On this criterion, the first requirement is to arrange the seating position so that the pilot's head and neck is normal to the applied acceleration, not only because of the need to look forward, but also to help the pilot to make head movements, under g. (Even so, under very high g, fighter pilots sometimes need to relax g momentarily, in order to make a head movement, to keep an opponent in sight).

There is also merit in raising the height of the rudder pedals on fighter aircraft. Rudder pedal movements are more easily made, and the blood circulatory system should have an easier task. In this way, the definition of a reclining seat begins (Figure 4). The angle α measures the tilt of the seat: an important angle from the designers point of view, of adjustment, ejection, and installation. From a physiological point of view, because the pilot's body is bowed, it becomes a less accurate measure of high g improvement.

Nevertheless, the question arises - what increase in g can be tolerated by increasing the angle α? Based purely on centrifuge tests, the answer is disappointing (Reference 2). A tilt of 30° shows negligible improvement, and by 60°, the increment is perhaps 1g, with further benefits beyond. Confirmation is contained in reference 3, from which figure 5 is derived. The relationship between the greyout threshold (no g suit) and the inclination of the aortic-retinal plane shows that 60° or more is needed for significant benefits. The fact of the matter is that, as we will see later, angles greater than 45° pose almost insurmountable design problems.

But the constant g centrifuge does not tell the whole story. Reference 4 indicates that the rate of onset of g affects the pilot's g threshold - a rapid onset (1g per second) reduces the black-out g by as much as 2, compared to a slow onset (0.1 g per second). In addition, tolerance to g is related to task environment, and incentive (it is easier to tolerate g if you are the pilot rather than the passenger). Increasing aircraft incidence with g acts in a favourable sense, and pilot opinion seems to reflect a benefit in g tolerance from tilt angles as low as 25°, if the heel line is also raised. A well-designed 30° seat will provide a greater degree of back and head support than a 15° seat, as illustrated by comparing figure 2 with 4. Perhaps comfort is as important a criterion as seat angle, in the design of a high g cockpit. Sustained manoeuvring at high g is a strenuous physical activity, and seat comfort will contribute to reduced pilot fatigue throughout the mission.

3. GEOMETRICAL FACTORS

Pilots come in all shapes and sizes. The likely variations are available to designers in publications such as references 5 and 6. An obvious design requirement is that sufficient room and sufficient adjustment to seat and controls exists, so that all pilots within say a 3-99 percentile of the anthropometric range shall be able to fly the aircraft, to reach all controls, and to seel all displays. The need for easy entry to and exit from the cockpit is also self-evident. These requirements apply to any aircraft. The design of a figher cockpit must respect additional needs. The first of these is concerned with escape, should the pilot need to abandon the aircraft in flight. Experience has established design criteria for safe ejection, including jettison of canopy, use of leg restraints, knee clearance of stick, panels and coaming.

The Military Specifications reflects this experience. The second (and all important) requirement for a fighter is that the pilot must have a suitable sighting system for the weapons which he intends to deliver. The head up display, HUD, fulfils this role, and has now evolved to provide primary flight reference.

The well-known advantages of this method of presentation are the flexibility of the format, the accuracy of the displayed information, and the positioning of the image against the outside world. The drawback from the cockpit designers point of view, is that the successful use of the HUD depends on accurate location of the pilot's eye position.

If we look at a typical current design (Figure 6) we can see how the combined requirements of sighting, ejection, instruments, stick and throttle location, secondary controls and selectors, constraint the designer. In this case, typical of many current aircraft, a 15° ejection line is assumed. Based on the same assumptions of the range or pilots to be accommodated, and the field of view which must be provided for the various displays, we find a remarkable similarity in the geometry of cockpits, from different design organisations, and different countries, even though the aerodynamic shape of the aircraft under comparison are radically different.

In the past, perhaps the main way in which the fighter aircraft designer has modified the cockpit design from that of the strike aircraft, is in providing a greater external field of view. Field of view is of prime importance in combat, particularly in the rear quadrant. The fuselage line, and the absence of canopy/windscreen support structure are vital factors, but in addition, if the pilot will accept a reduction in panel space, a lower sill line can be used (figure 7).

The compromise between external view and panel space becomes even more apparent in the design of a cockpit with a reclined seat. The pilot's eye position is determined by the HUD, so that inclination of the seat must be made relative to this datum. The angle between the seat back and seat squa cannot be much greater than 100°, if the problem of the pilot slipping through the seat during ejection is to be avoided. (Of course, developing a seat and harness which ejects in an upward direction would cure this problem, and create others). Ejection clearance dictates that as the seat inclines, so the distance from the pilots eye to coaming and panels also increases.

The panel space directly ahead of the pilot is also reduced (Fig. 8) with increasing seat tilt. The view over the nose of the aircraft must be maintained (say at 15°) and the pilot's lower limbs determine the bottom cut-off on the panels. By 45°, the panel space is very small indeed. This trend may be seen on fig. 9, which plots the front panel depth (excluding the central console between the pilots feet) for several current aircraft. By extrapolation a seat tilt of 45° removes all panel depth.

Fig. 8, may also be used to illustrate a further difficulty associated with the reclining seat. As the seat tilts so the need arises to relocate stick, controls, and selectors, to be within easy reach of the range of pilots covered by the requirement. Although such changes are possible, particularly in view of the advanced avionic technology now being applied to the cockpit, the problem is exaggerated at large inclinations by the difficulty which the pilot will experience in making lateral movements of his body. The viewing of side panels in current aircraft is assisted by such movements - fighter pilots in combat often fly with the top harness loosened, to assist the downward and rearward view - and it is clear that if the upper torso is supported by the seat, such movements are more difficult, particularly under 'g'. The choice of sill line is again critical. If it is wide enough to give ample display area, the external downward field of view will be restricted.

4. DISPLAYS AND CONTROLS

The loss of front panel area as the seat is tilted is not the tragedy that it might have seemed to be, twenty years ago. Simple electronic displays are already in use in aircraft cockpits - as indicators, and as alpha-numeric status devices; extending their use to other display requirements allows a radical re-thinking of cockpit design. At the same time, increasing confidence in computers, sensors and control technology makes electrical signalling of the primary controls, relaxed stability, and spin prevention an automatic choice for the new generation of fighter aircraft. With these techniques comes the opportunity to re-locate and re-design the primary flight controls, and to re-arrange the controls and displays accordingly.

4.1. Displays

Before replacing the electro-mechanical instruments by a multiplicity of CRT's, careful thought should be given to ensure that no existing facilities are lost. Conventional cockpits have gone through an evolutionary process, so that in spite of their imposing appearance (to the uninitiated) they enable pilots to do a series of complex tasks very effectively. Change for the sake of change is a bad principle. At the same time, the electronic displays have features which can be exploited to provide options which were previously not possible.

The evaluation of a new display format is better undertaken in a dynamic rather than a static environment. The more representative of the operational condition, the more valid the assessment, and the ground based simulator provides the best opportunities for such assessments. Several studies concerning the use of electronic displays in otherwise conventional cockpits have been made, relating both to military and civil use. Examples of such studies may be seen in references 7 and 8, and at Warton we have done similar work relating to low level strike missions.

Care must be taken in such tests to ensure that the displays to be evaluated are a practical proposition.

Is there sufficient space for the installation?

Are anti-vibration mountings necessary?

Does the electrical installation pose problems?

Will they be legible under all operating conditions?

What are the implications with respect to maintenance over the life of the aircraft?

These questions can better be addressed by the aircraft contractor, than by a research group specialising in display theory, since he has to hand the relevant experts, including pilots.

Pilot evaluation is the goal of such simulator tests. The primary measure of merit is the questionnaire, substantiated where possible by quantitative measures. One difficulty encountered with the use of quantitative measures is that of selecting parameters which reflect the improvements which are reported by the pilots. Another difficulty is the need to sift through the volumes of data which accumulate during the recording of missions, each lasting for perhaps an hour.

Broad principles concerning electronic displays which come out of such trials are

- electronic displays are acceptable to pilots, and produce better performance, due in part to their greater information content.
- multi moding is an effective way of utilising a limited display surface, particularly for navigational purposes, and for the monitoring of systems under normal conditions.
- pilots require certain classes of information at all times. If a failure occurs which requires immediate pilot action, it is unacceptable for the pilot to make control selections to diagnose the trouble.
- where possible, both analog format (dial or strip) and digital read out should be incorporated into the display.

The evaluation of displays for a high g cockpit is a more difficult exercise, because of the reduced panel space, and the large geometrical changes associated with such a cockpit. It is no longer a question of replacing conventional instrumentation by electronic equivalents; a completely new cockpit layout is required. The designer must ensure that all the controls and displays required by the pilot in any circumstance, from start-up to shut-down, are available. The starting point of such an exercise is to categorise all controls and displays, and to relate them to function, associated system, utilisation and redundancy. A comparison can then be made with the controls and displays which conventional cockpits contain. A basis for such a comparison is contained in reference 9.

The necessary controls and displays must then be allocated to the available panel area. Figure 10 illustrates the dramatic reduction in front panel area which is assoicated with a 45° seat. The side panels offer a useful alternative, but the need for multi-moding of display is evident.

Before pilot evaluation with active displays and with representative pilot tasks, cockpit space models must be built, and assessments are needed of the display formats in a non-active state. Individual displays can then be driven, followed by the full mission trials. Without such preparation, an expensive and time consuming assessment can be wasted, because the chosen display formats do not comply with the latest ideas relating to cockpit fit.

4.2. Controls

The conventional control stick invariably obscures the view of the lower central panel. In consequence, the displays and controls which are used only occasionally are located there - oxygen, engine start, for example. Figure 10 shows that the central area provides most of the display surface at large seat-tilt angles, and so there is a pressing need to re-locate the stick.

A comparison between 7 and 8 shows the changed disposition of the front and side panels, relative to the pilot. He can no longer reach the front panel or coaming in the advanced cockpit, and the position of the stick relative to fuselage structure is changed. It is no longer possible to arrange a fixed location of the stick which is within comfortable reach of the full range of pilots, when seat adjustment is made for the correct eye position.

Unless a datum adjustment to the stick is provided (as we do in the case of the rudder pedals), again there is a need to look for an alternative stick location. Electrically signalled controls gives the designer much greater freedom in the choice of the primary control stick. An installation on the right-hand side panel, such as the RAE hand controller described in reference 10, or the force side stick as used in the F-16 is one possibility. There are arguments both for and against force side sticks. Doubts remain as to whether pure force control is as good as a force/deflection feel system, particularly at low speed. There are also the needs to minimise the loss of side panel space (particularly if some form of arm support is needed) and to provide on the side controller all the switching functions which currently decorate the conventional stick top.

A novel alternative, which is being considered by BAC, is the use of a bifurcated central control (figure 11) or 'wishbone'. The control, attached to the seat, is a force and limited movement device, which can be used to fly the aircraft with either hand. Each limb of the stick is furnished with switches and selectors, and the possibility exists to incorporate engine control functions into the left-hand arm. The arrangement gives the pilot the maximum panel and console viewing area, because the stick only obscures the view of his knees/upper leg. Either hand may be used to operate devices on the side consoles. These include hand-controllers, weapon selection, devices to change configuration, jettison levers, communications, and the multiplicity of other functions over which the pilot has an element of control.

5. IMPLICATIONS

Simply stated, radical changes in fighter cockpit design are implied if the pilot's seat is tilted beyond 30°. A noticeable g alleviation, is not achieved until the seat is tilted beyond 45°, if the centrifuge evidence is the sole criterion. Variable geometry seats have been designed, which might provide the pilot with the best of both worlds, but they do not solve the cockpit designer's problems, they simply transfer his problems into a new area. Pilots will also be suspicious of articulated seats, until evidence is forthcoming that they do not decrease the probability of successful ejection.

We are therefore in a situation where the designer of a new fighter aircraft has a difficult decision to make. Several 1970 fighters - F14, F15, F18 - have high sustained 'g' capability, but do not resort to seat tilt for 'g' alleviation. The technology is available to design a high 'g' cockpit, but there are many questions to be asked, and the development costs are likely to be high. And the final question remains, is it worthwhile? To answer this question, research and development is needed on several fronts.

Perhaps the first item for investigation is to quantify the use that pilot's will make in combat of the advanced fighter's increased performance. Airborne trials are difficult and expensive, and cannot include the assessment of a new configuration. The combat simulator overcomes these difficulties, but is deficient in other ways. The pilot does not experience the 'g' (and so his tactics are not modified by physical discomfort), and the fights are usually limited to 1 on 1. Even so, valuable data can be obtained from the simulator. Our experience in simulating fights of current modern fighter performance (maximum sustained turn rate 18°/second) has shown the need to use high 'g' - mean 'g' during a three minute fight greater than 5. The tactics and maneouvres in a dog fight are influenced by the weapons which each fighter carries, and we have also seen a trend, when simulating more maneouvrable aircraft (maximum sustained turn rate 25°/second) for the speed of the fight to decrease, and for the mean 'g' to reduce dramatically. The high turn rate opponent with a high performance short range missile rapidly forces the pilot to reduce speed, and to go for minimum turn radius. The maximum sustained turn rate is no longer the dominant criterion; in any case, at speeds where turn rate is not influenced by the structural limit, the maximum sustained turn rate has a relative flat optimum with speed. The combat simulator fights can be extended to study the tactical options - slashing attacks, breakaway manoeuvres, low level penetration - and the benefit that might derive if the pilot's 'g' tolerance could

be increased.

Confirmation of combat simulator trends can sometimes be obtained from flights, but perhaps the biggest contribution that flight trials can make is to examine physiological effects which have been investigated at Aero-Medical Centres, on centrifuges and other instruments of torture. For example, flight evaluation under controlled conditions are need of articulated seats, before a decision could be made to fit them to a production aircraft.

Section 6 briefly discussed the process of display and control evaluation. The type of display and controls is dependent on the emphasis that the specification puts on missions other than visual air-to-air combat. It could be argued that during air to air combat, the pilot should have no need to refer to information within the cockpit.

Some of our simulation research activity is directed towards the display of information during combat. The HUD and helmet-mounted displays are regarded as the primary sources of such information. It is however, difficult to resist the lure of the electronic cockpit - the designer is in the position of Alladdin (or is it Ali-Baba) - and somehow these sparkling displays have to be integrated into a working system. In one sense, the small available display area favours the designer. Display developments in the past have been additive - additional dials, indicators, HUD - but now the opportunity is there to reduce the number of active elements. The integration of the control and display functions with the airborne computer is part of this process. The difficulties should not be under-estimated - even in laboratory conditions, the interface problems are imposing; the additional airborne requirements concerning integrity, size, weight, and operational environment add to the development programme. Nevertheless, the need for a detailed study of the new options is clear - even if we find that the benefits to be obtained do not relate directly to providing the pilot with an ability to withstand greater 'g' forces.

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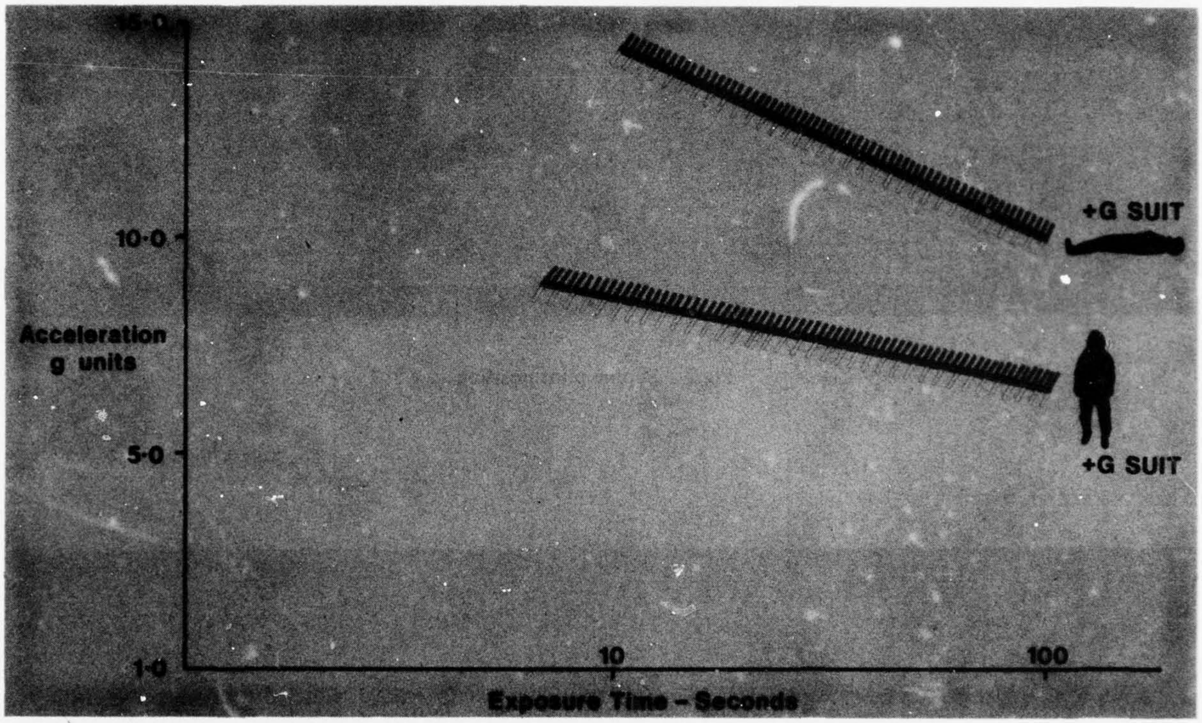

Fig.1 Tolerance to 'G' blackout threshold

Fig.2 Supine pilot position

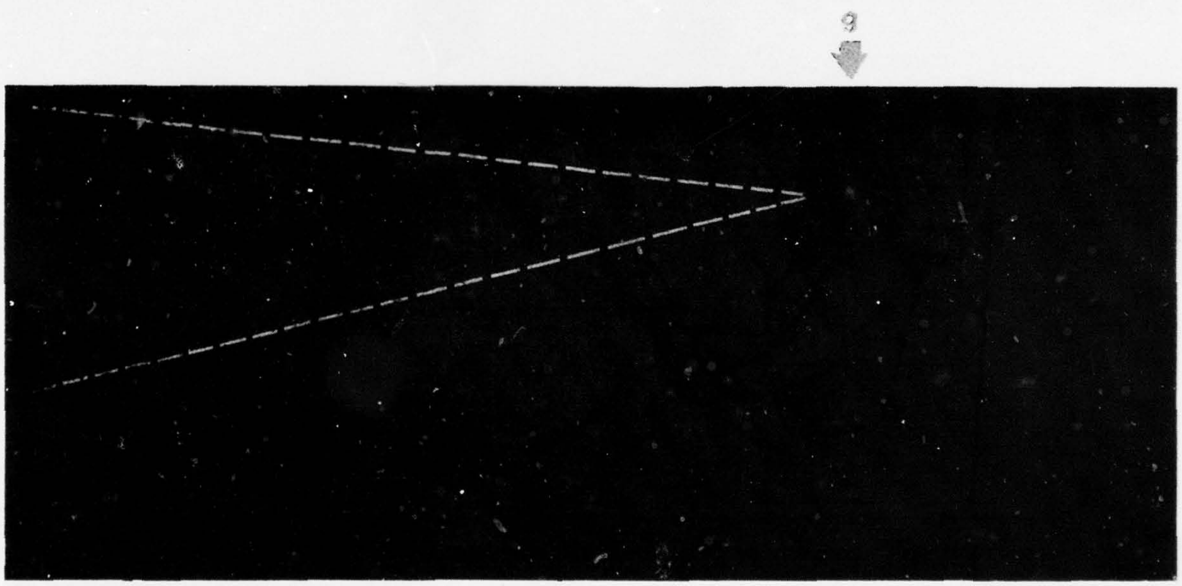

Fig.3 Conventional pilot position

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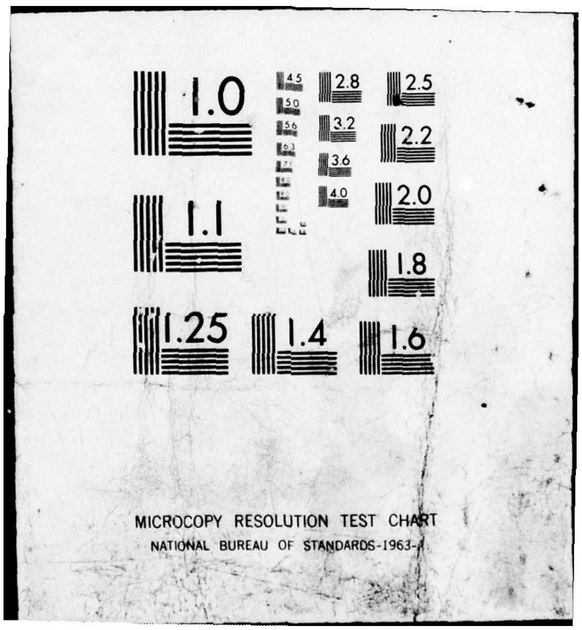
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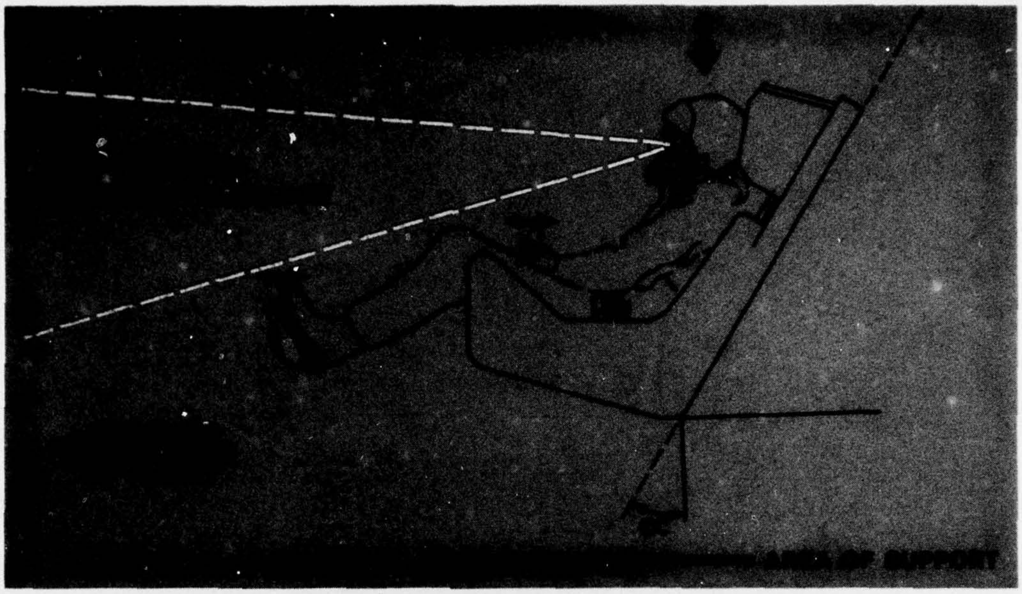

Fig.4 Reclined seat

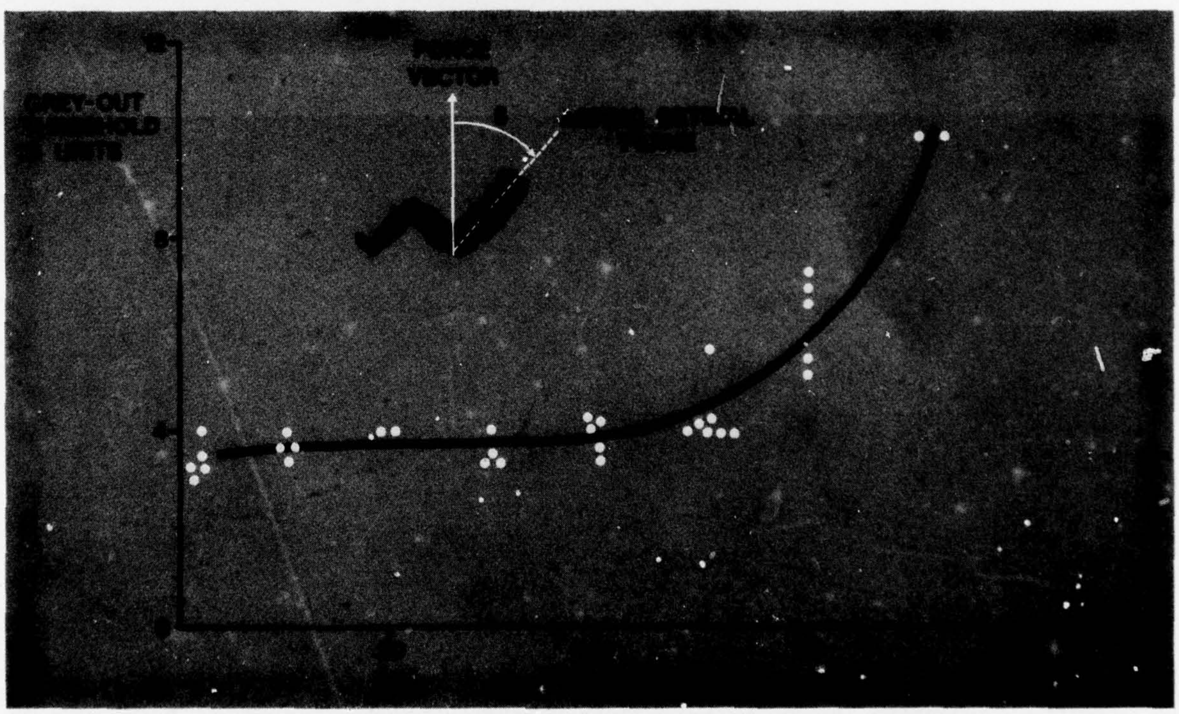

Fig.5 Influence of tilt on G tolerance

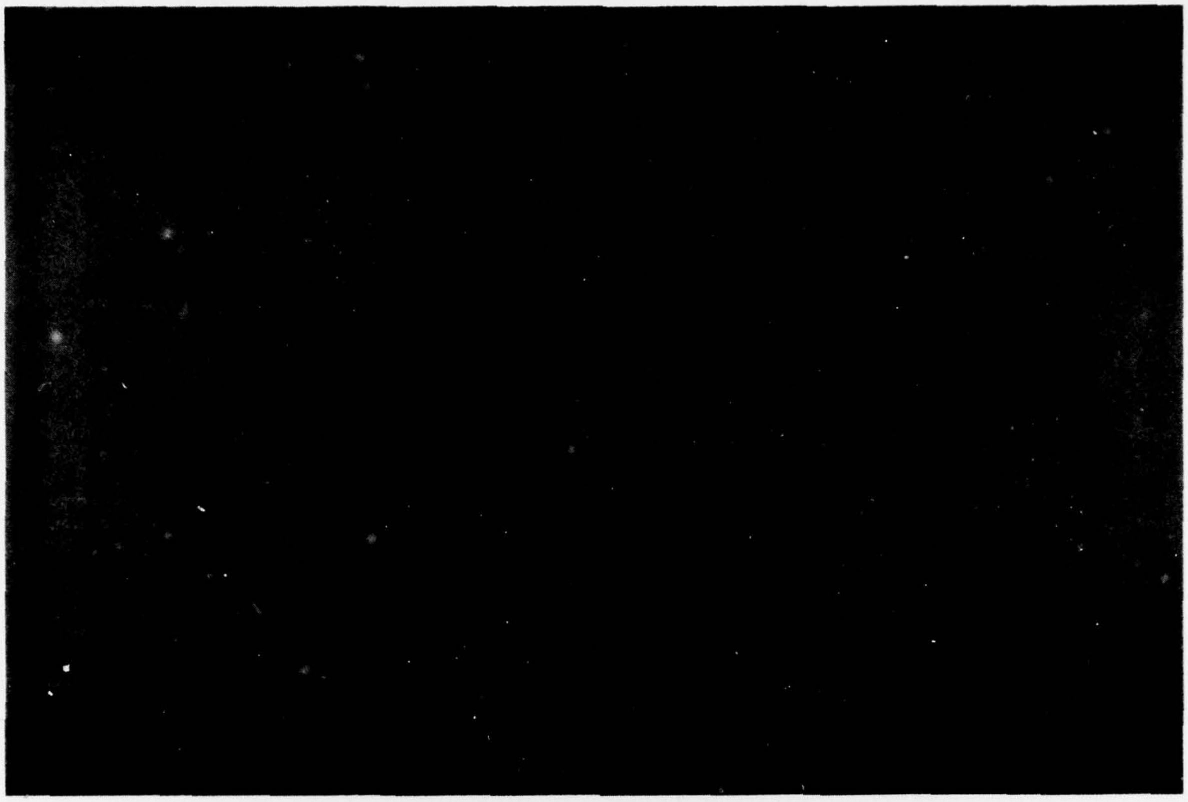

Fig.6 Conventional cockpit

Fig.7 'Fighter' cockpit

Fig.8 Advanced cockpit

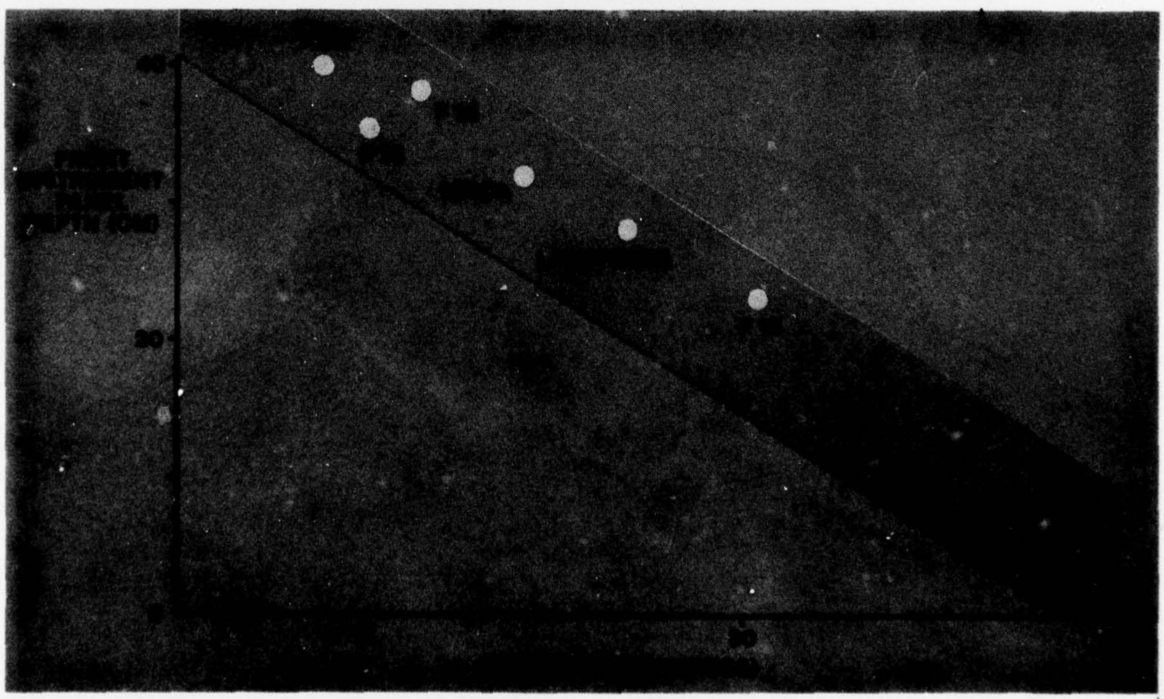

Fig.9 Influence of seat tilt on panel space

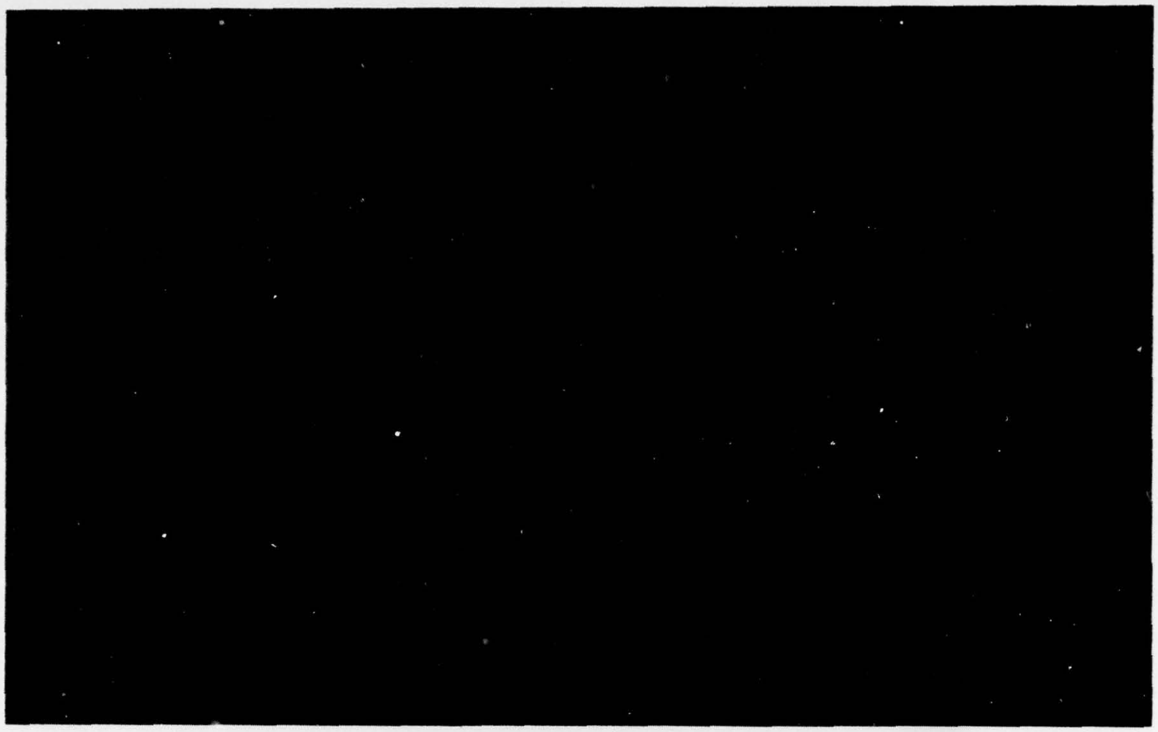

Fig.10 Front panel areas

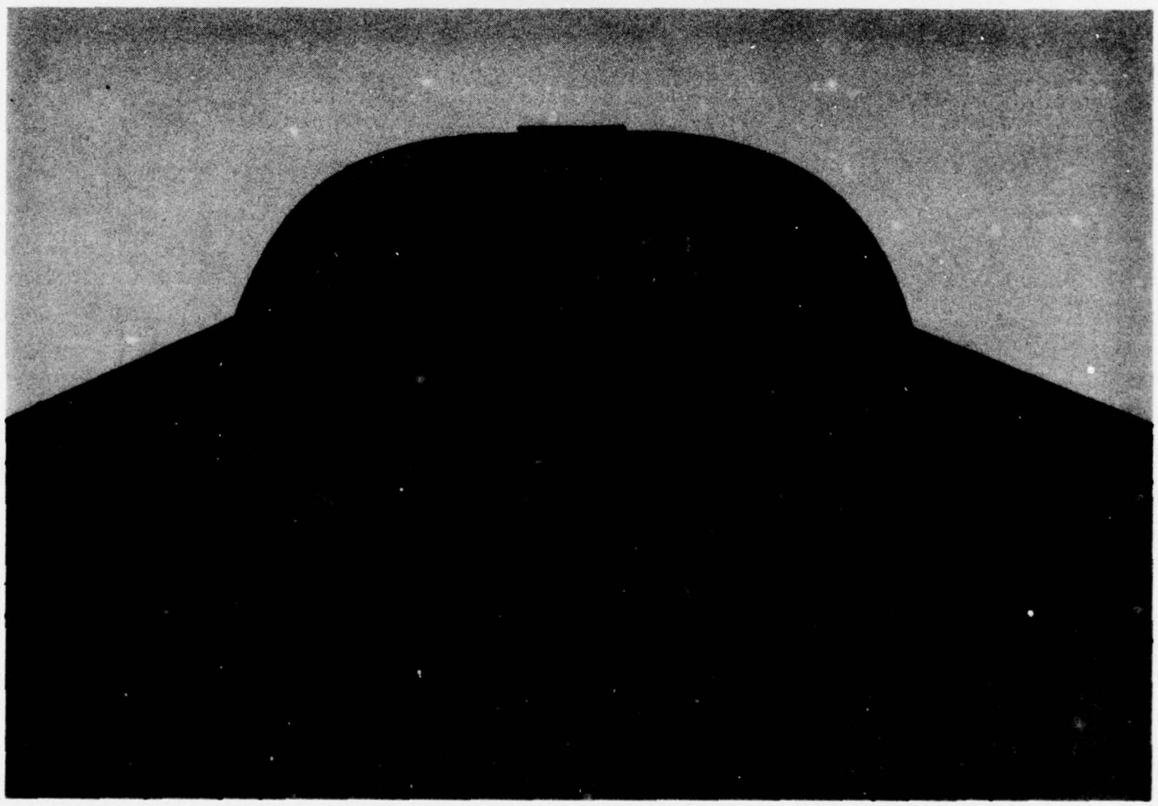

Fig.11 Bifurcated stick

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